CONCEPTUAL DESIGN STUDY OF
1985 COMMERCIAL VTOL TRANSPORTS
THAT UTILIZE ROTORS

Summary

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16. Abstract

Conceptual design studies of pure and compound helicopter commercial short-haul transport aircraft for initial fabrication in 1980 were performed to determine their technical and economic feasibility. One-hundred-passenger configurations were optimized for minimum direct operating cost consistent with producibility and marketability, with emphasis on proper account of mass properties, performance and handling qualities adequacy, and suppression of internal and external noise. The effect of external noise constraints was assessed, in terms of gross weight and direct operating cost, for each aircraft.

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FOREWORD

Sikorsky Aircraft, a Division of United Aircraft Corporation, has conducted a study entitled "Conceptual Design Study of 1985 Commercial Transports That Utilize Rotors," under Contract NAS2-8079 from the National Aeronautics and Space Administration, Ames facility. The study was conducted between February and August, 1974. NASA technical representatives were Mr. Gary Churchill and Mr. Demo Giulianetti. The authors wish to acknowledge their assistance as well as that provided by the following Sikorsky personnel in the disciplines indicated:

K. C. Hansen	Handling Qualities
D. K. Unsworth	Mass Properties
S. A. Schmidt	Performance
J. W. Jones	Economics
A. C. Whyte	Aircraft Design
R. W. Beckert	Aircraft Design

The results of the study are presented herein as Volume I of two volumes. Volume II, NASA report CR-137598, contains the technical substantiation for these results.

SUMMARY

The objective of this study was to design the largest size helicopter and compound commercial transports that would be feasible and practical if fabrication would begin in 1980, to a maximum of 100-passenger capacity, as constrained by an external noise restraint to be evaluated. The effect of a variation of this noise restraint on the design and operation of these aircraft was then assessed. Handling qualities, payload, and mission capability were similar throughout.

The selected external noise criterion was 95 PNdB at a 150-meter (500 foot) sideline in hover on a sea level 32.2-degree C (90-degree F) day. Payload was set at the study guideline maximum of 100 passengers, considered feasible and practical in terms of size for the defined timeframe of initial fabrication in 1980. Baseline optimization was generally achieved by minimizing direct operation cost (DOC),using the Aerospace Industries Associates (AIA) cost model, over a 370-kilometer (200-nautical mile) stagelength.

A twenty-five percent saving in structural weight from current state-of-the-art trending was assumed, representative of the use of composite materials. It was also assumed that current knowledge in noise reduction techniques for main rotors can be applied to tail rotors. A 5 dB reduction in external noise signature for a given turboshaft engine size was assumed for improvement in compressor design techniques, within the prescribed timeframe.

The most significant result was that the helicopter achieves the noise limit goal with no compromise to optimum selection of rotor parameters. The compound, when constrained by study guidelines of constant rotor geometry, is compromised in that the low blade twist and low blade area desirable for high speed flight are not consistent with low noise in hover. Helicopter DOC was 4% lower than that of the compound and showed smaller increases at reduced range. Helicopter gross weight was 26,371 kilograms (58,137 pounds), cruise speed 89 meters/second (173 knots). Compound gross weight was 34,440 kilograms (75,926 pounds), cruise speed 128.6 meters/second (250 knots). From examination of DOC and noise sensitivities around the baseline, it was possible to select rotor parameters to achieve the ±5 PNdB members of the two families, considering minimum change to DOC. The quiet helicopter was achieved through reduction in rotor tip speeds and the adoption of twin low-disc-loading tail rotors, for 4% increase in DOC. The quiet compound was achieved through reduction in rotor tip speeds and adoption of a fan-in-fin in place of a conventional tail rotor, for 6% increase in DOC. PNL contours during a take-off procedure show somewhat greater enclosed areas at a given noise level for the compound than for the helicopter. This is primarily because the compound must follow a flatter take-off profile than the helicopter, using auxiliary propulsion to avoid negative wing lift and/or high vertical drag penalties.

The results of this study are expected to form part of a general broad-based analysis of all VTOL concepts. The baseline DOCs of 1.973 cents per seat kilometer (3.174 cents per seat statute mile) for the helicopter and 2.051 cents per seat kilometer (3.30 cents per seat statute mile) for the

compound are about 20% above those for current fixed wing shorthaul commercial aircraft of similar size. However, through use of small city-center V-ports, the VTOL aircraft offers the business traveler substantial reduction in access cost and time, and will divert air traffic from congested CTOL facilities. In suburban localities, the VTOL machine presents the opportunity to move more people per unit time per unit of terminal area than either CTOL or STOL aircraft, because many simultaneous landing and take-off operations can take place. This independence from prescribed runways eliminates the problems of traffic holding, either on the ground or in the air, typical of today's CTOL airports. The VTOL aircraft, therefore, represents a competitive and highly marketable mode of transportation when compared with existing inter-city systems.

	HELICOPTER	COMPOUND
GROSS WEIGHT, kg	26371 (58,137 lb)	34440 (75,926 lb)
WEIGHT EMPTY, kg	15592 (34,374 lb)	22482 (49,564 lb)
PASSENGERS	100	100
ROTOR DIAMETER, m	28.1 (92.2 ft)	26.9 (88.4 ft)
ROTOR DISC LOADING, kg/m^2	41.5 (8.5 psf)	58.7 (12 psf)
INSTALLED POWER, mhp	10753 (10,605 hp)	22287 (21,979 hp)
WING LOADING, kg/m^2	-	418 (85.5 psf)
V CRUISE, M/sec	89 (173 kt)	129 (250 kt)
CRUISE ALTITUDE, m	1219 (4000 ft)	4267 (14000 ft)
AUX.PROP.	-	PROP-FANS
FLIGHT CONTROLS	FLY-BY-WIRE	FLY-BY-WIRE
BODY STYLE	6 - ABREAST SINGLE AISLE	6 - ABREAST SINGLE AISLE
HOVER TIPSPEED, m/sec	222.5 (730 fps)	210.3 (690 fps)
EXTERNAL NOISE, PNdB (150-meter sideline)	93.5	95
INTERNAL NOISE, PSIL	70	70
95 PNdB FOOTPRINT AREA, km^2		
TAKEOFF	.195 (.075 sq. mi)	.405 (.156 sq. mi)
LANDING	.163 (.063 sq. mi)	.227 (.088 sq. mi)
BLOCK FUEL, kg	1544 (3404 lb)	2440 (5379 lb)
BLOCK TIME, hr	1.331	.958
DOC, ¢/seat km:		
370 km (200 n.m.)	1.973 (3.174¢/seat mile)	2.051 (3.30 ¢/seat mi)
740 km (400 n.m.), pass.	2.153 (3.464¢/seat mile), 83	2.428 (3.906¢/seat mi) 74

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LIST OF SYMBOLS

AIA	Aerospace Industries Associates
A_n	Acceleration normal to aircraft longitudinal axis, m/sec^2
A_{1_s}	Main rotor lateral cyclic blade pitch angle measured in shaft axis system, cosine term of Fourier series representation of blade pitch angle, positive for stick right, deg.
a_{fp}	Acceleration along the aircraft flight path, m/sec^2
B	Baseline
B_{1_s}	Main rotor longitudinal cyclic blade pitch angle measured in the shaft axis system, sine term of Fourier series representation of blade pitch angle, positive for stick forward, deg.
b	Number of rotor blades
C	Compound
CARD	Civil Aviation Research and Development
CTOL	Conventional Take-off or Landing
C_T/σ	Main rotor thrust coefficient in the shaft axis system, positive up, $T/\rho\pi R^2(\Omega R)^2 \sigma$
c	Blade chord, m
DOC	Direct Operating Cost, cents/seat-Km
DL	Main rotor disc loading, $GW/\pi R^2$
db	Decibel
dbA	Decibel A-weighted
EPNL	Effective Perceived Noise Level
FH	Flight-Hour
FSCG	Fuselage Station of Center of Gravity, cm
GW	Aircraft Gross Weight, kg
g	Acceleration due to gravity
H	Helicopter
HP	Horsepower - U.S.
H_z	Hertz frequency, cycles/sec
HLH	Heavy Lift Helicopter
I_{xx}	Mass moment of inertia of aircraft about the principal fuselage longitudinal axis without main rotor blades, $kg-cm-sec^2$
I_{xz}	Cross product of inertia, $kg-cm-sec^2$
I_{yy}	Mass moment of inertia of aircraft about the principal fuselage lateral axis without main rotor blades, $kg-cm-sec^2$

I_{zz}	Mass moment of inertia of aircraft about the principal vertical axis without main rotor blades, kg-cm-sec^2
i_T	Horizontal tail incidence, degrees
K_g	Fuselage wing-body gust alleviation factor
Kgr	Rotor gust alleviation factor
L_{DN}	Day-night noise level
$\mathcal{L}$	Fuselage rolling moment
$\mathcal{m}$	Fuselage pitching moment
MH	Man-hour
mHP	Horsepower - metric
N	Noisy
$\mathcal{n}$	Fuselage yawing moment
N_z	Aircraft load factor, g's
OEI	One engine inoperative
OGE	Out-of-ground effect
PNdB	Perceived Noise Level, decibels
PNL	Perceived Noise Level
p	Fuselage roll rate - angular velocity, rad/sec
Q	Quiet
q	Fuselage pitch rate - angular velocity, rad/sec
$\dot{q}$	Fuselage pitch acceleration - angular acceleration, rad/sec^2
R	Rotor radius, m
RSRA	Rotor System Research Aircraft
r	Fuselage yaw rate - angular velocity, rad/sec
SAS	Stability Augmentation System
SENEL	Single Event Noise Energy Level
T	Rotor thrust, kg
T_J	Single propulsor thrust, kg
TBO	Time between overhauls
TF	Thrust factor
U_{de}	Derived gust velocity, m/sec
V	Aircraft forward speed, m/sec
V/STOL	Vertical/Short Take-off or Landing
VTOL	Vertical Take-off or Landing

V_{BR}	Best Range Speed, m/sec
V_x	Component of forward speed along the fuselage, x-axis, m/sec
V_y	Component of forward speed along the fuselage, y-axis, m/sec
V_z	Component of forward speed along the fuselage, z-axis, m/sec
Z/D	Ratio of fuselage/propulsor clearance to propulsor diameter
α_s	Main rotor shaft incidence angle, degrees
β_B	Body sideslip angle, degrees
δ_3	Rotor blade pitch-flap coupling angle, degrees
ζ	Damping ratio
ρ	Mass density of air
σ	Rotor solidity ratio, $bc/\pi R$
θ	Fuselage pitch angle, degrees
θ_B	Fuselage pitch rate, degrees/sec
θ_{mR}	Main rotor blade collective pitch at the center of rotation
θ_{TR}	Tail rotor blade pitch at .75R
ϕ_B	Fuselage roll attitude
φ_B	Fuselage yaw attitude
Ω	Rotor angular velocity
$\overline{\Omega}$	Rotor tipspeed ratio Ω/Ω_o
Ω_o	Rotor angular velocity at design speed
ω_η	Undamped natural frequency

1.0 INTRODUCTION

A VTOL air transportation system operating into the demand centers of air travel offers a possible solution for the problem of choked air facilities. With its excellent maneuverability at low speeds and its ability to hover, a VTOL aircraft can use small city-center V-ports, thereby promising reduction of access cost and time for passengers and diversion of air traffic from congested CTOL airports. It also enables more effective use of suburban facilities because of accelerated passenger rate per unit time per unit of terrain area used, and freedom from prolonged traffic holdings. Such a system will offer the airline a city-to-city network with passenger appeal superior to that of the equivalent CTOL system. The primary advantage of the VTOL system is reduced trip time, which is of predominant importance to business travelers.

To enable meaningful comparison of competitive VTOL concepts, it is desirable that each should represent reasonable technological goals to be achieved in a given timeframe, and that each should reflect equivalent technology advance from the current state of the art. For transportation systems, operational costs generally decrease with vehicle size, or passenger payload. Establishment, then, of the maximum viable size of each concept technologically feasible within the given timeframe has a fundamental bearing on the results of a transportation systems study. The primary objective of this study was to perform a conceptual design of helicopter (Figure 1-1) and compound helicopter (Figure 1-2) transports, of a size considered technologically feasible for initial fabrication in 1980.

Previous general VTOL transportation system studies have analyzed route structures and compared the operational economics of each concept, but have tended to ignore the environment effects of external noise. Yet noise level is a primary concern to VTOL operators in downtown areas. Containment of noise within acceptable limits will dictate selection of lift and propulsion system components and may require special powerplant noise suppression equipment and constraints on operational techniques. It is essential, therefore, to assess the effects of noise level regulations on commercial VTOL aircraft design. A meaningful noise limit criterion must be established to assure that the aircraft is not unduly compromised by the designer in attempting to conform with an overly rigorous or unrepresentative regulation. An initial task in this study was the selection of such an external noise restraint as a design groundrule for the subsequent study.

The secondary objective of this study was to parametrically determine the effect of a selected noise criterion on helicopter and compound designs, in particular on aircraft gross weight, performance, direct operating cost and technical risk. Families of these two VTOL aircraft were derived, each consisting of a baseline design plus two designs constrained by noise levels above and below the selected noise criterion, but otherwise offering the same handling qualities, payload, and mission capability. In addition, compliance with the community acceptance criteria established in a recently completed study by Sikorsky for NASA/Langley, Reference 1, was assessed.
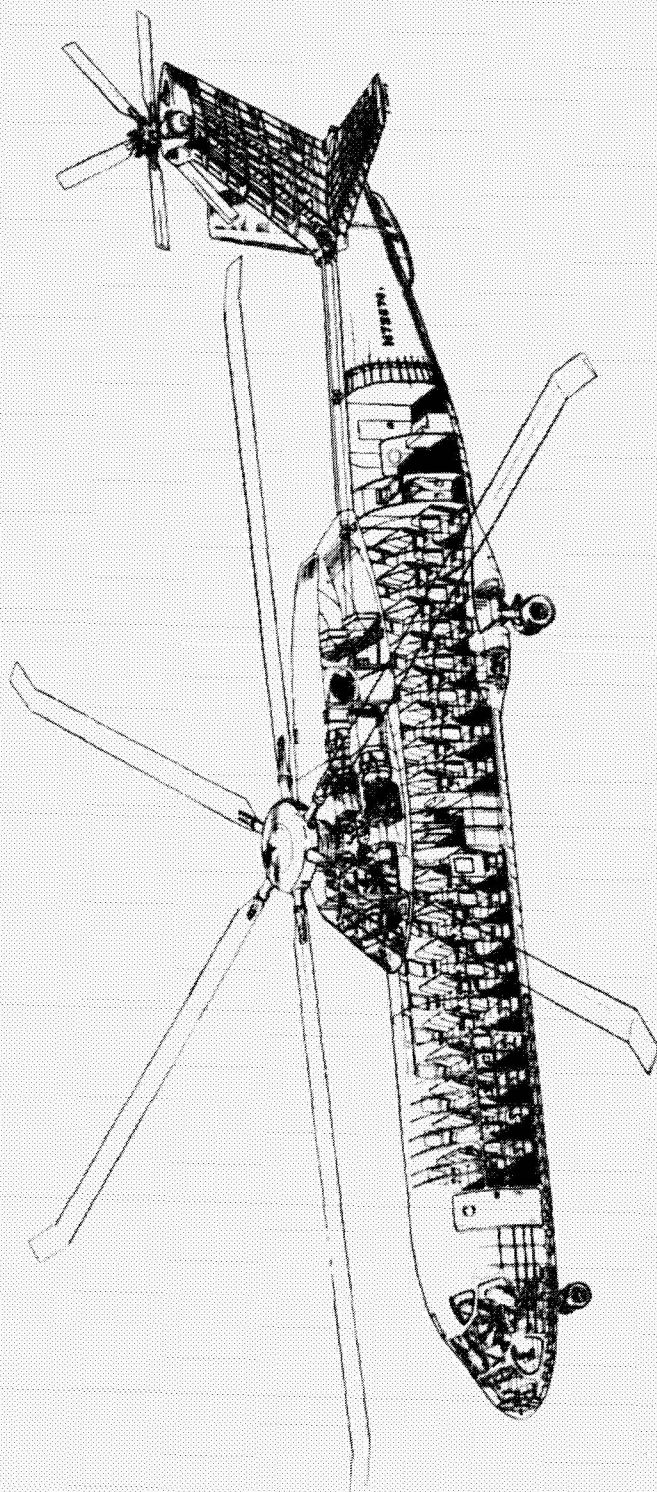
FIGURE 1-1. 100-PASSENGER COMMERCIAL HELICOPTER ISOMETRIC
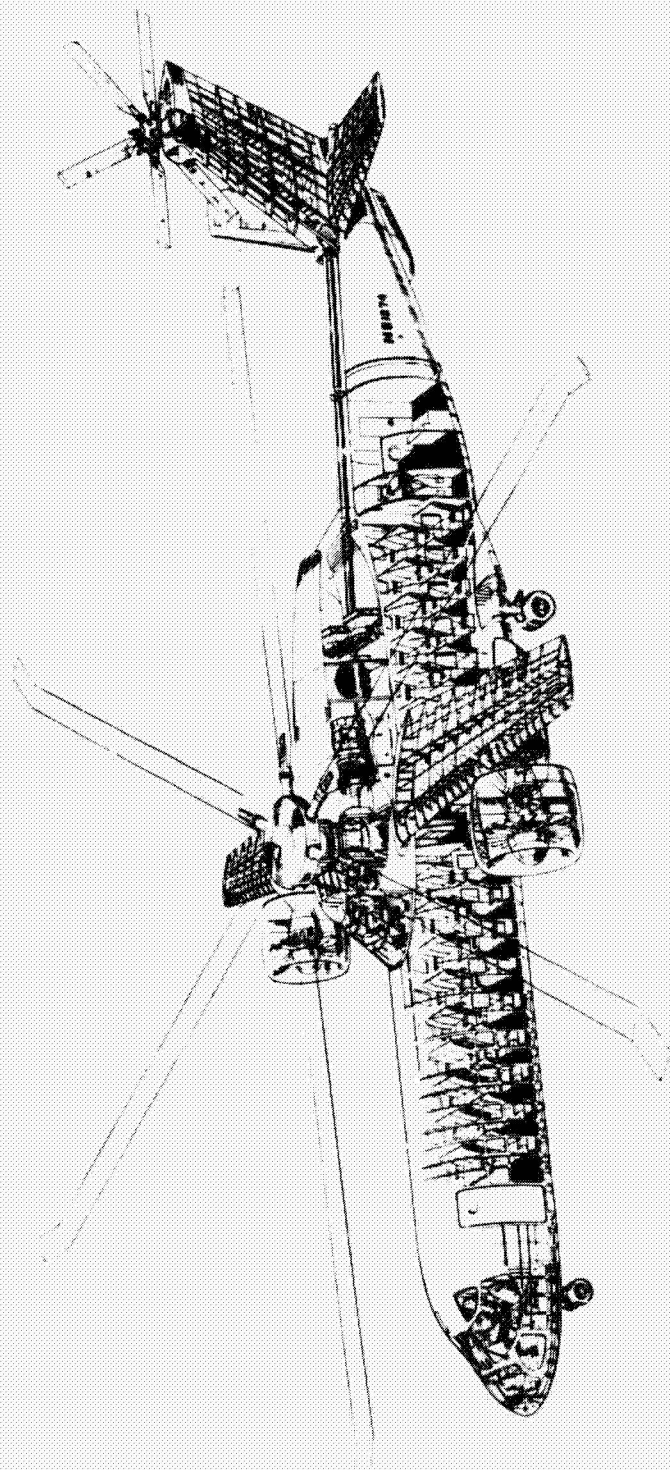
FIGURE 1-2. 100-PASSENGER COMMERCIAL COMPOUND ISOMETRIC

2.0 DESIGN GUIDELINES AND ASSUMPTIONS

For each VTOL concept, a related family of three aircraft was required, each designed to a different external noise limit level. To facilitate identification of each design, the following nomenclature was adopted:

Configuration:

H - Helicopter
C - Compound

Qualifier:

B - Baseline
Q - Quiet
N - Noisy

The level of effort expended in performing the sensitivity and trade-off portions of the stvdy (QH, NH, QC, and NC designs) was approximately equal to the effort expended in determining the BH and BC designs. Thus the baseline aircraft are defined in greater detail than the other members of each related family. The study sequence is summarized as follows:

- review study guidelines (Reference 2)

- assess technological risk as a function of aircraft size (payload volume)

- establish acoustic analysis methodology

- derive BH and BC designs as required to minimize DOC, and establish sensitivity of DOC to changes 'in major design parameters

- establish baseline external noise levels and sensitivity of noise level to major design parameters

- adjust BH and BC designs, as appropriate, to achieve baseline noise goals at minimum possible DOC

- select sets of rotor parameters as predicted from DOC and noise sensitivity analysis, in order to achieve QH, NH, QC, and NC designs with minimum DOC for these particular noise criteria

The primary study guidelines are listed in Figure 2-1. The first consideration for deriving the baseline aircraft was to minimize Direct Operating Cost (DOC). The selected baseline external noise limit at a 150 meter (500 foot) sideline was 95 PNdB. This is appropriate because (1) it has been suggested as a possible certification level, and (2) it enables compliance with the community acceptance criteria at most typical heliport locations considered for the studies recently completed under contract from NASA/Langley, Reference (1). The cabin internal speech interference level in cruise of 70dB PSIL is consistent with current fixed-wing jet design practice. The guideline of fixed rotor geometry precluded consideration of variable twist and variable diameter compound concepts in this study. (It is believed that either of these innovations could significantly reduce compound DOC, because they provide

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compatibility of hover performance and low noise requirements with low rotor
drag characteristics in cruise flight.) Primary assumptions are listed in
Figure 2-2.

PASSENGERS	100 MAXIMUM
STAGELENGTH	200 N.M.
V CRUISE	MINIMUM DOC *
HOVER	OUT OF GROUND EFFECT, ONE ENGINE INOPERATIVE, @ SEA LEVEL 32.2°C(90°F)
INITIAL FABRICATION	1980 (INTRODUCTION TO SERVICE IN 1985)
EXTERNAL NOISE	95 PNdB 500-ft SIDELINE
INTERNAL NOISE	70 PSIL IN CRUISE
CABIN VIBRATION	.05 g
ROTOR	FIXED GEOMETRY
AIRCRAFT OPTIMIZATION	MINIMUM DOC
CRUISE ALTITUDE	MINIMUM DOC
* OR OTHER CONSIDERATION AS APPROPRIATE	

FIGURE 2-1. PRIMARY DESIGN GUIDELINES

WEIGHTS:
. 25% OF STRUCTURAL WEIGHT SAVING THROUGH USE OF COMPOSITE
  MATERIALS

PERFORMANCE:
. HELICOPTER ROTOR PERFORMANCE TRENDED FROM YUH-60A UTTAS
  DESIGN POINT - EXTRAPOLATED INTO HIGH SPEED COMPOUND REGIME
  BY GENERALIZED ROTOR PERFORMANCE (GRP) METHOD (REFERENCES 4 AND 5)
. ENGINE PERFORMANCE AND HP/LB BASED ON ALLISON 501-M62
. HAMILTON STANDARD Q-FAN AND OPTIMIZED PROPELLER DATA

ECONOMICS:
. AIA COST MODEL (SEE FIGURE 2-4)

EQUIPMENT:
. AS FOR EASTERN AIRLINES NORTH-EAST CORRIDOR STUDY (REFERENCE 6)

ACOUSTICS:
. 10 dB INTERNAL NOISE REDUCTION FROM TRANSMISSION ISOLATION
. TAIL ROTOR NOISE REDUCTION FOLLOWS CURRENT MAIN ROTOR TRENDS

FIGURE 2-2. PRIMARY DESIGN ASSUMPTIONS

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2.1 Mission and Economics

Figure 2-3 shows the helicopter and compound mission profiles and cruise altitudes selected for the baseline designs. Distances traveled during acceleration, climb, and descent are credited toward the 370 kilometer (200 nautical mile) stagelength. Climb and descent rates were limited to 2.54 m/sec (500 fpm) and 1.52 m/sec (300 fpm) for the helicopter, for which the cabin is not pressurized.
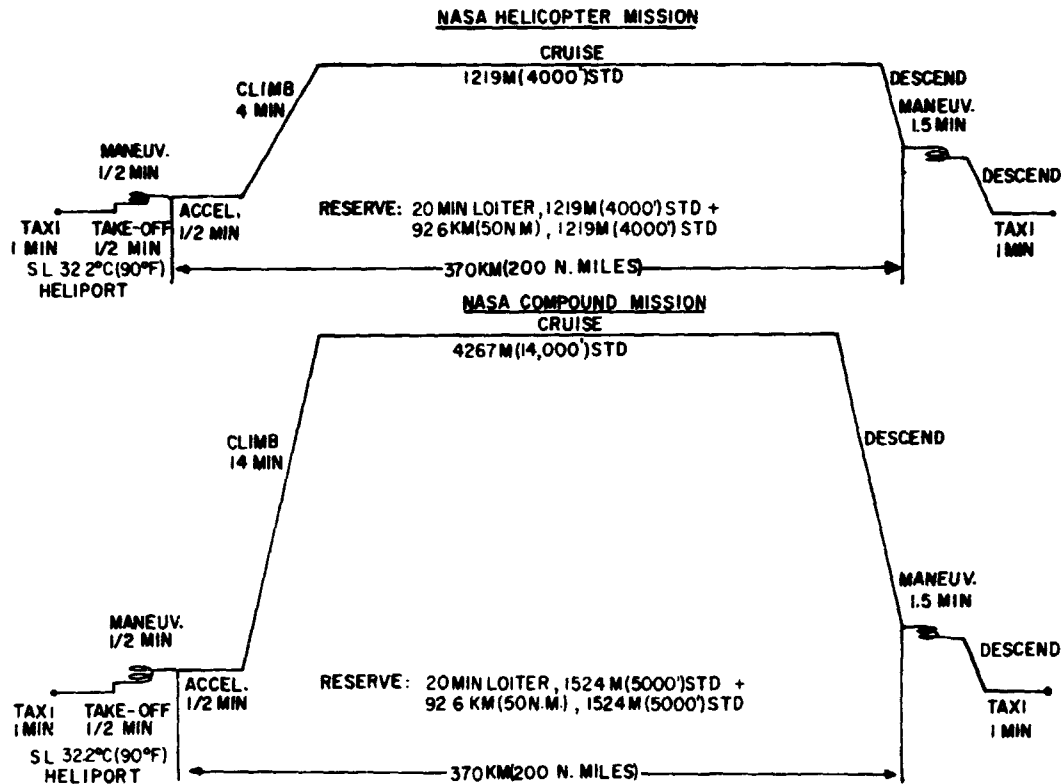
FIGURE 2-3. HELICOPTER AND COMPOUND MISSION PROFILES

The Aerospace Industries Associates (AIA) cost model was used to compute DOC. This method for evaluating direct operating cost (Reference (3)) was developed in 1968 by Aerospace Industry representatives with coordination by the Vertical Lift Aircraft Council of the Aerospace Industries Association, Inc. Where appropriate, other factors, such as fuel economy, vehicle productivity, or design feasibility, were used in preference to the absolute minimum DOC point. Figure 2-4 shows the assumptions made for input to the AIA cost model. Airframe price and vehicle utilization rate were trended from the baseline values shown.

YEAR DOLLARS	1974
AVIONICS PRICE - $/A/C	250,000
AIRFRAME PRICE - $/LB	110 (TRENDED)
DYNAMIC SYSTEM PRICE - $/LB	80
ENGINE PRICE - $/RATED SHP	280 ($HP^{.785}$)
CREW COSTS - $/HR	$\frac{.067\ GW}{1000} + 134$
FUEL - $/LB	.02
OIL - $/LB	1.24
NONREVENUE FACTOR	2%
LABOR RATE - $/HR	6.0
AIRFRAME LABOR - MH/FH	1.0 AIA
AIRFRAME MATERIAL - $/FH	1.0 AIA
ENGINE LABOR - MH/FH	.65 AIA
ENGINE MATERIAL - $/FH	.65 AIA
ENGINE TBO-HR	4500
DYNAMIC SYSTEM LABOR - MH/FH	AIA
DYNAMIC SYSTEM MATERIAL - $/FH	AIA
DYNAMIC SYSTEM TBO-HR	3000
MAINTENANCE BURDEN	150% DIRECT LABOR
DEPRECIATION PERIOD - YEARS	12
SPARES	
AIRFRAME	8
ENGINES	40
DYNAMIC SYSTEM	25
UTILIZATION - HOURS	2500 (TRENDED)

FIGURE 2-4. ECONOMICS ASSUMPTIONS

2.2 Noise

2.2.1 Internal Noise

Study guidelines dictate that internal noise is to be no higher than 70 dB in the Preferred Speech Interference Level (PSIL) throughout the cabin during cruise, and no more than 75 dB PSIL during takeoff. These requirements have a primary effect on aircraft design, necessitating proper transmission acoustic isolation, cabin wall soundproofing and, in the case of the compound, careful selection of auxiliary propulsion.

Helicopter internal noise in the Speech Interference Level region is almost exclusively controlled by noise generated by the main transmission. This occurs at gear meshing frequencies and is primarily pure tone noise. As yet, the analytical ability to accurately predict transmission noise has not

been perfected. However, Figure 2-5 shows how bare cabin PSIL generally trends with installed power. For preliminary design purposes this curve, derived from measured aircraft data, can be used with confidence.
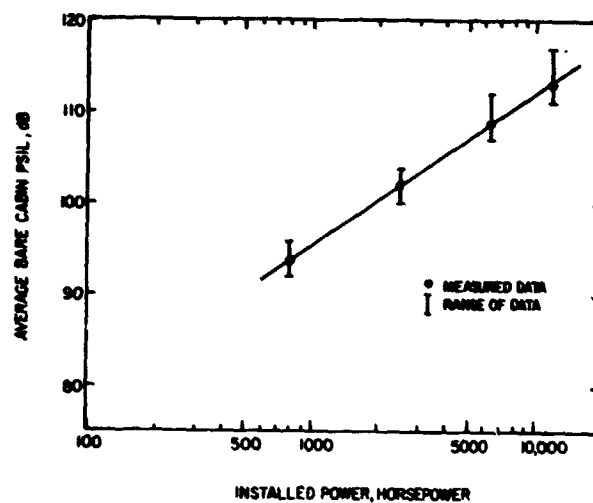
FIGURE 2-5.   HELICOPTER BARE CABIN SPEECH INTERFERENCE
LEVELS AS A FUNCTION OF INSTALLED POWER

The soundproofing required to meet the specified noise levels must generally be designed in detail in order to account for all noise sources and to minimize acoustic leakage. For preliminary design purposes, it is possible to use a generalized trending curve developed from measured helicopter noise data. Figure 2-6 shows this trend based on current (1972) commercial design techniques and on advanced (1976-1980) technology design techniques. The advanced technology curve is based on laboratory tests of newer materials and techniques. This includes integral trim and acoustic panels utilizing floating septums in open cell foams and effective acoustic isolation of all panels from the airframe. It is obvious that if PSIL reductions on the order of 30-40 dB are required, very heavy soundproofing will be necessary.

The required soundproofing weight can be reduced by treating the primary source of the noise, the main transmission.

Reduction in cabin noise (PSIL) of about 23 dB can be achieved by an accumulation of the following design techniques:

(a) Acoustically phased planetary gear sets (about 7 dB).

(b) Damping of larger spur and bevel gears (about 6 dB).

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(c) Transmission isolation at acoustic frequencies (about 10 dB).
These methods have been tested in practice, as discussed in References 7 and 8.
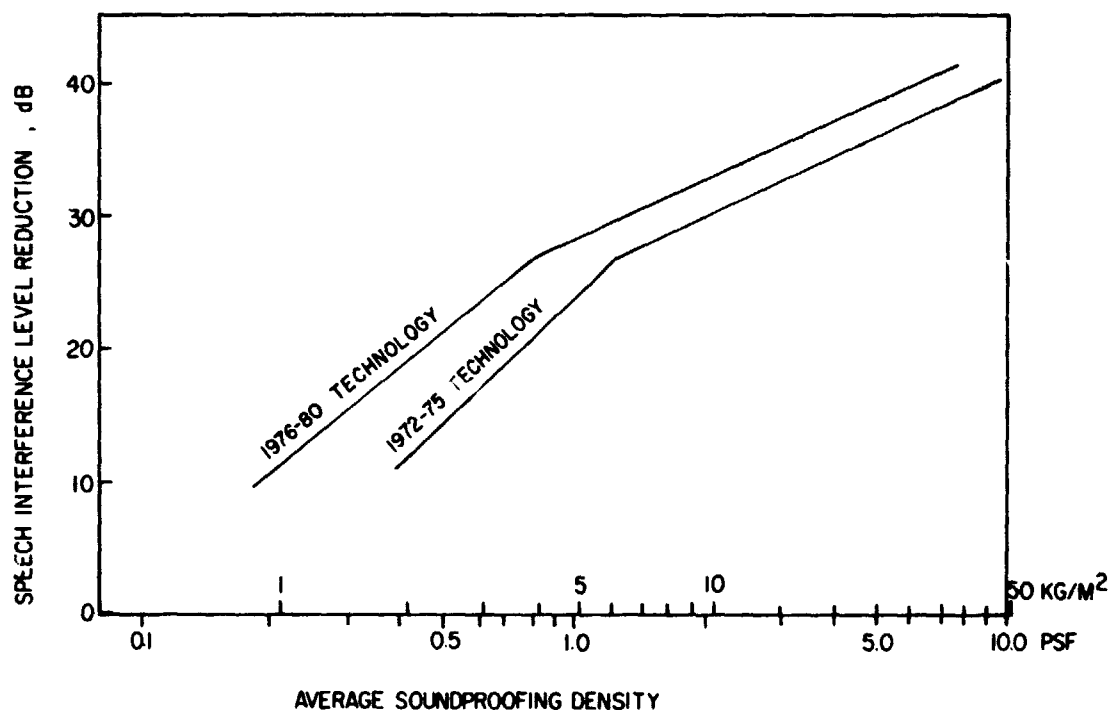
FIGURE 2-6. COMMERCIAL SOUNDPROOFING EFFICIENCY

2.2.2 External Noise

The selected external noise design requirement is that the noise level
150 meters (500 feet) to the side of the baseline aircraft should not exceed
95 PNdB. The quiet and noisy designs were to be approximately 5 PNdB quieter
and noisier, respectively. In addition, a comparison was undertaken in this
study of the aircraft noise with community acceptance guidelines developed by
Munch and King in Reference 1. This criterion involves all factors present
in a typical civil operation, such as aircraft noise duration and spectral con-
tent, time of day, type of neighborhood, and number of operations a day.

The assumption is made that 1980 technology components will be available
for the vehicles under consideration. For the turboshaft engines, this means
a noise level approximately 5 PNdB lower than a current engine of comparable
horsepower. This assumption is based on the results of the NASA Quiet Engine
Program and the results of a joint DOT-NASA Civil Aviation Research and Develop-
ment Study (CARD study) postulating a reduction of 10 dB a decade in engine

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noise. Also, Reference 9 states that modest technology advances are expected to lower core engine noise by 5 PNdB by 1980.

For rotors, 1980 technology means use of advanced rotor geometry, employing sophisticated airfoils, special twist distribution, and new tip designs. Many of these concepts have already been tested, and confidence is high in the ability to predict their noise accurately.

2.3 Stability and Control

The helicopter and compound helicopters were designed to meet the specific requirements of Appendix A of Reference 2 as amended by the guideline review coordination meeting held at NASA/Ames on February 11 and 12, 1974. Reference 10 also was considered consistent with the requirements of Reference 2, quantitative dynamic stability and flying qualities analysis were conducted.

2.4 Technology

2.4.1 Size

In order that the results of this study can be used in conjunction with other VTOL design studies, it is desirable that each design represent comparable advances in materials and component size in relation to previously manufactured hardware. Technological advances must result in products that can be manufactured at a reasonable cost, on a reasonable schedule, and with acceptable risk. Component size can be regarded as one facet of technological advance. Historically, gross weight growth above a factor of about 2.5 times the largest gross weight aircraft of a similar type previously built has resulted in significant production overweight above the predicted value. This has been contributed by unforeseen manufacturing difficulties in fabricating large pieces of hardware, weight penalties to overcome unforeseen development problems, and additional fuel because of optimism in predicting forward flight performance.

The problem is really not one of scaling alone. The larger vehicle might optimize at a different rotor configuration (different rotor disc loading, solidity, blade number, blade twist, etc.), perhaps beyond the range of parameter combinations for which experience exists.

Performance and noise requirements dictate that the scaling up in size must necessarily occur at a fairly constant blade tip Mach number. For constant disc loading, the square-cube relationship would predict intolerable increases in rotor and drive system weight. Also, constant disc loading would impose large penalties on fuselage length and weight, in the case of a single-rotored or tandem-rotored aircraft. In the case of a side-by-side rotor configuration, the rotor size affects the wing span and weight. It follows then, that a scaling up of rotorcraft at the same blade tip Mach number requires higher rotor disc loading. Blade loading is limited by adding blades and/or by adding chord for the same number of blades (lowering aspect ratio). For example, in the case of the scaling up from the XH-51A with 3 blades and 1583-kilogram (3500-lb) gross weight to the AH-56A with 4 blades and 8301 kilograms

(18,300 lb) gross weight, the rotor diameter was increased from 10.67 m (35 ft) to 15.61 m (51.2 ft) and the rotor disc loading from 17.57 (3.6) to 43.43 kg/sq m (8.9 lb/sq ft). This was only possible with a much higher blade solidity ratio. The resulting much lower blade aspect ratio of the AH-56A was one of the reasons why the scaled up gyro-control system did not perform as well as in the XH-51A. As this example shows, a 5-to-1 scaling up of the gross weight from a previously largest rotorcraft of similar configuration will result in radically different rotor geometry, with as much as twice the blade solidity and rotor disc loading. This magnifies any problem related to rotor downwash impingement.

Aeroelastic problems of the scaled-up rotorcraft will be quite different from those of a much smaller design. Because of the lower blade aspect ratio or higher blade solidity ratio, the blade Locke number will increase. The nature of any aeroelastic problem varies significantly with blade Locke number.

From the point of view of vibration control, the scaled-up rotorcraft will operate entirely outside the spectrum of a much smaller rotorcraft design. The rotor rpm will be much lower, so that the rotor modes will have correspondingly lower frequencies. With respect to vibrations, it is exceedingly difficult to avoid wing resonances over the wide operational rpm range of an in-flight variable rpm rotor system.

The CH-53E, with a 24.08-meter (79-foot) rotor diameter and 11509 metric horsepower (11,350 horsepower) gearbox, has already flown at more than 31,750 kilograms (70,000 pounds) gross weight. It was therefore considered that the commercial helicopter payload size should be the 100-passenger limit imposed by the study groundrules, with little or no technical risk. For the compound, the implications of a 34,000-kilogram (75,000 pound) aircraft were not as well defined. Although the compound does not include any single innovative lift system element, the wing/rotor combination has been operated at high speed only on a relatively small prototype aircraft, such as the 8618-kilogram (19,000-pound) NH-3. By 1976, the Rotor Systems Research Aircraft (RSRA) will be flying as a 13,154-kilogram (29,000-pound) compound helicopter, providing a thorough understanding of compound flight up to 300 knots. This aircraft would provide a scale factor of about 2.6 to a 34,000-kilogram (75,000-pound) 100-passenger commercial compound, considered a justifiable technology advance without high risk.

Extrapolation of a configuration to larger size is usually possible, technically. That is, no fundamental laws prevent the development. However, the larger the extrapolation, the greater the uncertainty of the predicted weights and performance of the resultant aircraft. Weight penalties for solving unknown problems cannot be estimated - in fact, in preliminary design there is a tendency to ignore these penalties.

A statistical method was derived to create a weight empty contingency function varying with gross weight growth factor. This factor is representative of the growth, in terms of gross weight, of the conceptual design under study, compared with the largest aircraft previously built of a similar

configuration. The method of Reference 11 was used, based on a large number of manufactured helicopters. The resulting curve is shown in Figure 2-7. This relationship, which was added to the weight trending section of the helicopter design computer model, can be used to predict weight contingencies for other configurations of VTOL aircraft. Thus, VTOL configurations of similar passenger capacity can be meaningfully compared, with their respective DOCs adjusted for technological risk by way of the weight contingency. This method predicted a weight contingency of about 0.12% of weight empty for the study baseline helicopter, and about 1.6% of weight empty for the compound.
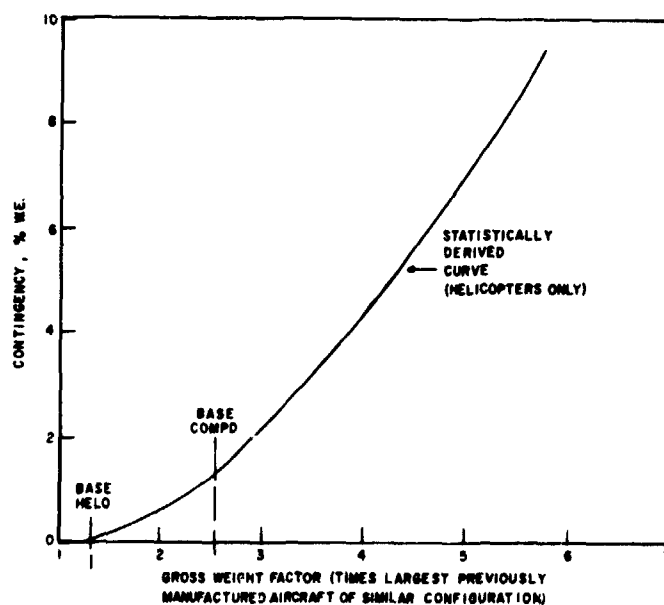
FIGURE 2-7. WEIGHT EMPTY CONTINGENCY ASSESSMENT FOR TECHNOLOGICAL RISK ASSOCIATED WITH SIZE

12

2.4.2 Performance

Performance evaluation was based on standard methodology, References 4 and 5, with technology level adjusted for a production aircraft in 1985.

Aerodynamic performance efficiencies in the rotor and drive system are representative of the U.S. Army UTTAS helicopter program, with 1% improvement in hover figure of merit for technology advance in blade design. Drag estimates were based on manufactured hardware data and/or win   inel test results.

Engine performance was scaled from a selected basel?   verplant.   This baseline is the Allison 501-M62 model, which matches the   cific fuel consumption requirements for this study and is in the size range required.   It is also representative of the required production timeframe.   The baseline engine performance characteristics are shown in Figure 2-8.

Auxiliary propulsion performance for the compound aircraft is based on published data for the Hamilton Standard Q-fan^{TM} concept which has been demonstrated under test conditions, References 12 and 13.
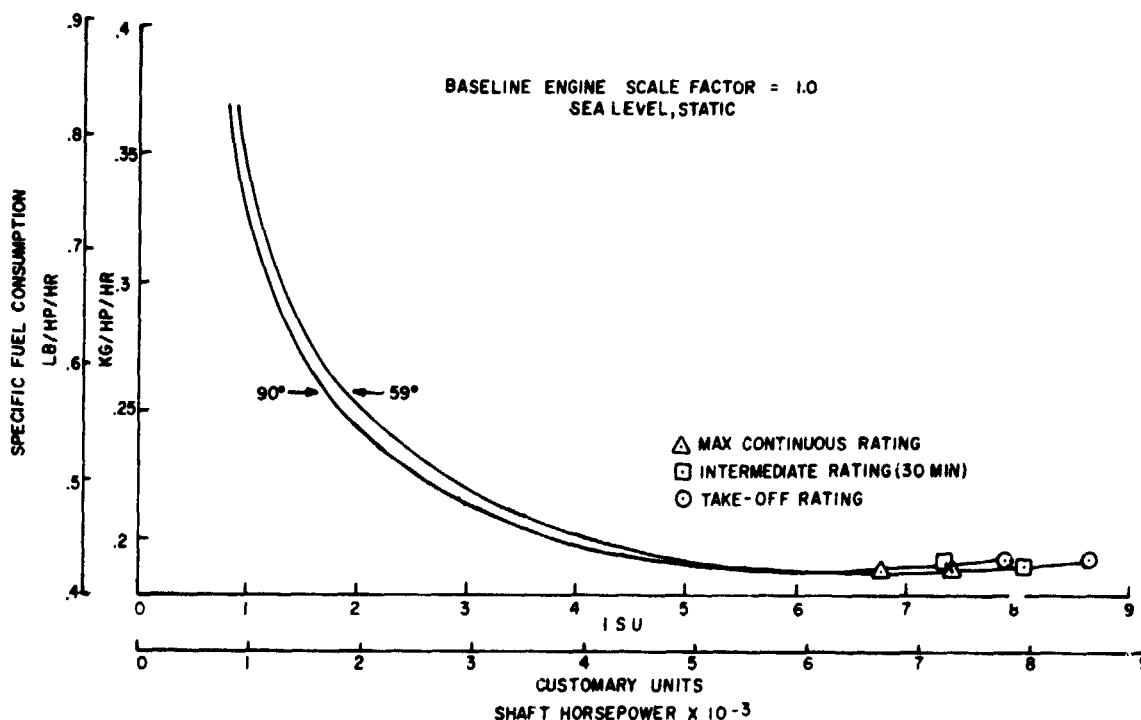
FIGURE 2-8. BASELINE ENGINE CHARACTERISTICS

GROUP	% REDUCTION IN WEIGHT DUE TO TECHNOLOGICAL ADVANCEMENT	MATERIALS TECHNOLOGY REQUIRED TO MEET TECHNOLOGICAL ADVANCEMENT
AIRFRAME	25	All-composite primary structure of graphite epoxy. Secondary structure of kevlar or graphite epoxy.
WiNG	25	All-composite primary structure of graphite epoxy. Control surfaces of graphite epoxy. Secondary structure of kevlar or graphite epoxy.
ENGINE SECTION	11	Cowling of graphite epoxy design, mounts and fire-walls of high strength steel or titanium, fairings of kevlar or graphite epoxy.
FLIGHT CONTROLS	20	Complete fly-by-wire system, no mechanical back-up.
ELECTRICAL	14	Oil cooled generators (current technology but not included in statistics). Introduction of teflon wire along with further minia-turization of relays, etc.
VIBRATION SUPPRESSION	10	Improvement in design techniques.

FIGURE 2-9. PERCENTAGE REDUCTION IN COMPONENT WEIGHTS
DUE TO ADVANCED TECHNOLOGY

2.4.3 Mass Properties

The gross weights of the helicopter and compound baseline designs were estimated taking into account the technological advances in structure, controls, and equipment that should be available by the early nineteen-eighties. The percentage reduction in component weights and a brief description of the materials technology required to achieve these weight reductions are shown in Figure 2-9. These percentage weight reductions were taken in agreement with NASA.

The effects of advances in technology on the gross weight of the two baseline designs are shown in Figure 2-10. In this figure, a current technology solution to the baseline design would have a gross weight 10% higher for the helicopter and 12% higher for the compound. A 1985 solution, when compared with the baseline, would have a gross weight 4% less for the helicopter and 5% less for the compound. These 1985 technology solutions take advantage of forseeable weight saving techniques for that timeframe.
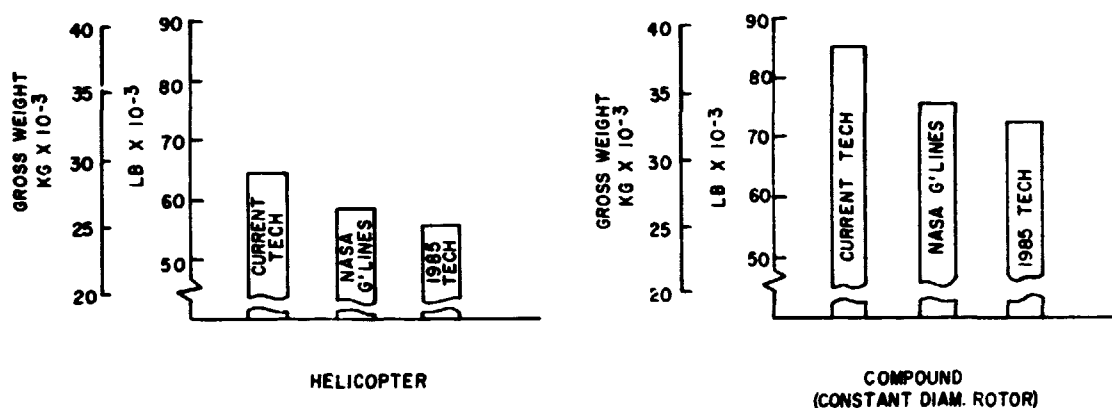
FIGURE 2-10.   EFFECTS OF ADVANCED TECHNOLOGY ON GROSS WEIGHT

15

3.0 BASELINE DESIGNS

3.1 Aircraft Description

Figure 3-1 summarizes the primary attributes of the baseline helicopter and compound designs. Figures 3-2 and 3-3 are three-views of the two aircraft, described in detail in Figure 3-4.

	HELICOPTER	COMPOUND
GROSS WEIGHT, kg	26371 (58,137 lb)	34440 (75,926 lb)
WEIGHT EMPTY, kg	15592 (34,374 lb)	22482 (49,564 lb)
PASSENGERS	100	100
ROTOR DIAMETER, m	28.1 (92.2 ft)	26.9 (88.4 ft)
ROTOR DISC LOADING, kg/m²	41.5 (8.5 psf)	58.7 (12 psf)
INSTALLED POWER, mhp	10753 (10,605 hp)	22287 (21,979 hp)
WING LOADING, kg/m²	-	418 (85.5 psf)
V CRUISE, M/sec	89 (173 kt)	129 (250 kt)
CRUISE ALTITUDE, m	1219 (4000 ft)	4267 (14000 ft)
AUX.PROP.	-	PROP-FANS
FLIGHT CONTROLS	FLY-BY-WIRE	FLY-BY-WIRE
BODY STYLE	6 - ABREAST SINGLE AISLE	6 - ABREAST SINGLE AISLE
HOVER TIPSPEED, m/sec	222.5 (730 fps)	210.3 (690 fps)
EXTERNAL NOISE, PNdB (150-meter sideline)	93.5	95
INTERNAL NOISE, PSIL	70	70
95 PNdB FOOTPRINT AREA, km²		
TAKEOFF	.195 (.075 sq. mi)	.405 (.156 sq. mi)
LANDING	.163 (.063 sq. mi)	.227 (.088 sq. mi)
BLOCK FUEL, kg	1544 (3404 lb)	2440 (5379 lb)
BLOCK TIME, hr	1.331	.958
DCC, ¢/seat km:		
370 km (200 n.m.)	1.973 (3.174¢/seat mile)	2.051 (3.30 ¢/seat mi)
740 km (400 n.m.),	2.153 (3.464¢/seat mile),	2.428 (3.906¢/seat mi)
pass.	83	74

FIGURE 3-1   BASELINE AIRCRAFT PRIMARY ATTRIBUTES

16
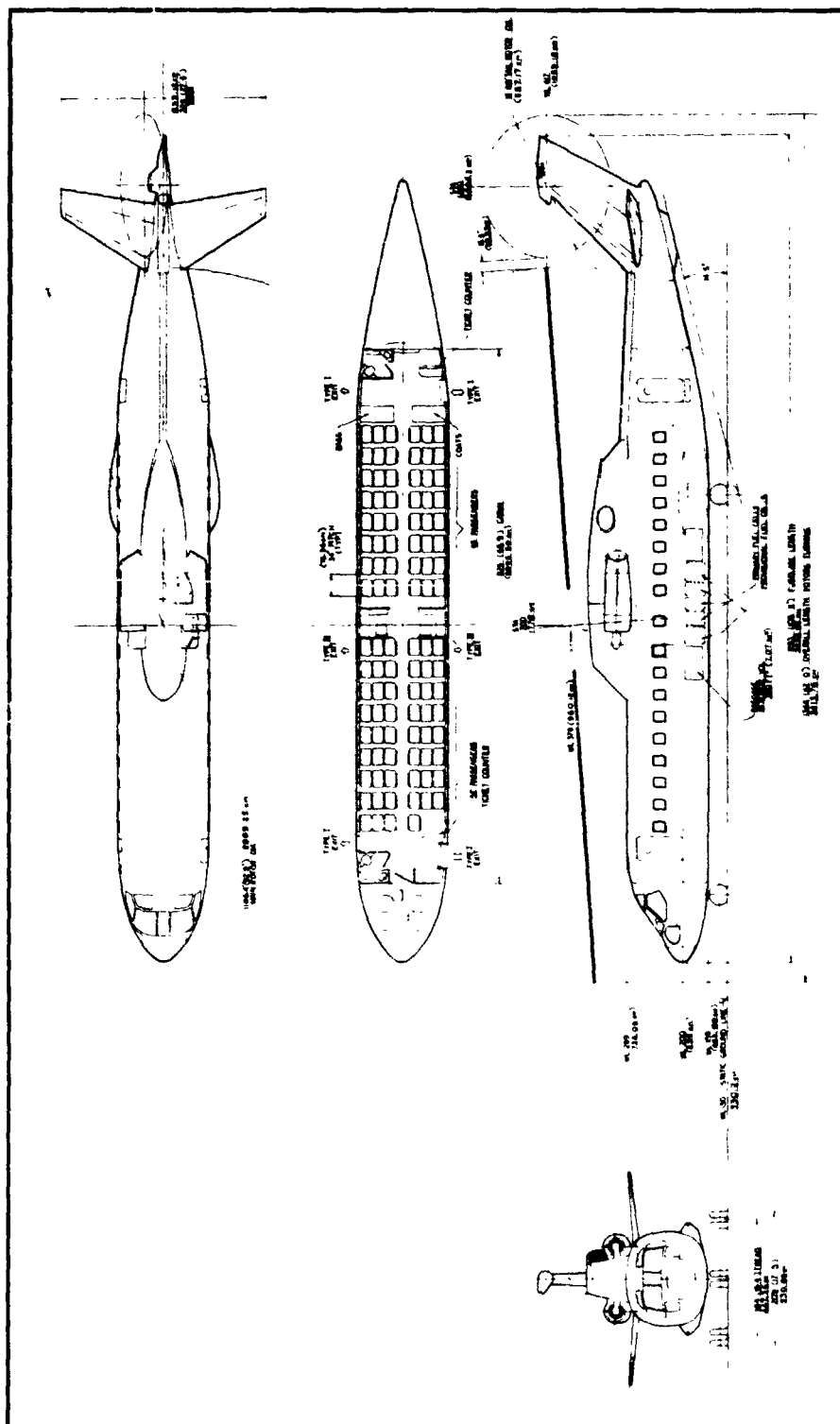
FIGURE 3-2. BASELINE HELICOPTER 3-VIEW

17
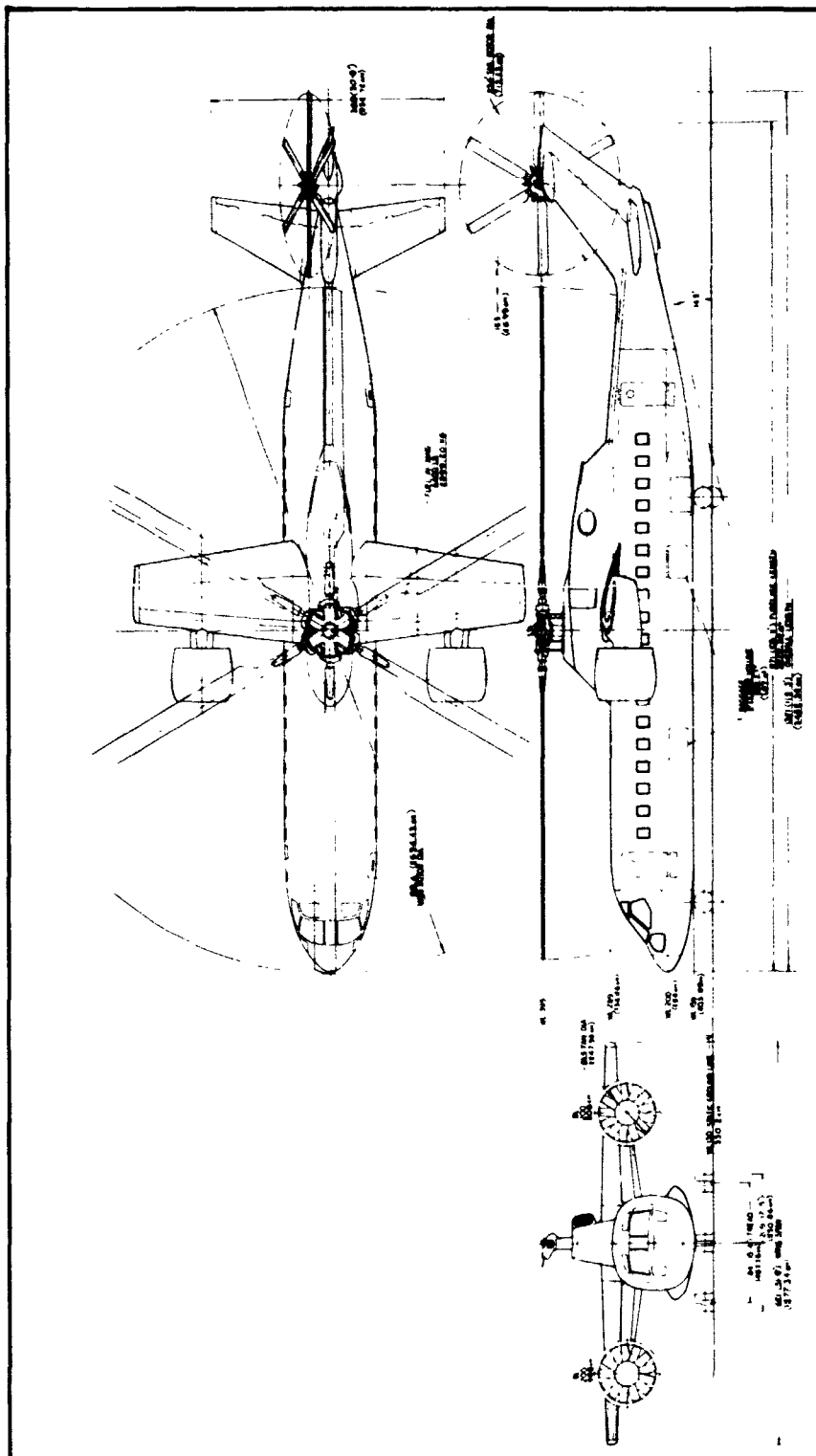
FIGURE 3-3. BASELINE COMPOUND 3-VIEW

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	Helicopter	Compound
Main Rotor		
Type	Articulated	Articulated
	Lubricated hinges	Lubricated hinges
	Swept tips	Swept tips
	Faired head	83 pitch flap coupling
		faired head
Airfoil Section	SC1095	SC1095
Diameter, m	28.10 (92.2 ft)	26.94 (88.4 ft)
Chord, cm	72.19 (28.42 in.)	85.52 (33.67 in.)
Blades	5	6
Hover tipspeed, m/sec	222.5 (730 fps)	210.3 (690 fps)
Cruise tipspeed, m/sec	222.5 (730 fps)	158.50 (520 fps)
Hinge offset, %	6.33	6.55
Equivalent linear twist, deg	-16	-12
Hover C_T/σ @ SL 90°F	.075	.1
Shaft tilt, deg	-3	0
Vibration suppression	bifilar	bifilar
Wing		
Span, m	-	15.81 (51.86 ft)
Aspect Ratio	-	4.75
Airfoil section	-	23015
Area, m²	-	52.58 (566 ft²)
Mean aerodynamic chord, m	-	170.2 (67 in.)
Devices	-	Semi-span flaps, leading edge spoiler
Tail Rotor		
Type	Crossbeam	Crossbeam
Diameter, m	5.58 (18.3 ft)	7.22 (23.7 ft)
Chord, cm	27.46 (10.81 in.)	38.56 (15.18 in.)
Blades	6	6
Hover tipspeed, m/sec	213.4 (700 fps)	210.3 (690 fps)
Equivalent linear twist, deg	-16	-12
Cant, deg	20	20
Tail Surfaces		
Style	Inverted T	Inverted T
Horizontal area, m²	17.28 (186 ft²)	23.23 (250 ft²)
Vertical area, m²	14.31 (154 ft²)	14.31 (154 ft²)
Body		
Cross Section	Double ellipse	Double ellipse
Wetted area, m²	310.76 (3345 ft²)	306.58 (3300 ft²)
Cabin length, m	21.01 (68.92 ft)	21.01 (68.92 ft)
Seat abreast	6	6
Aisles	1	1
Pressurization, newton/cm²	None	4.137 (6 psi)
Maximum width, cm	371 (146 in.)	371 (146 in.)
Maximum height, cm	330 (130 in.)	330 (130 in.)
Overall length, m	32.59 (106.9 ft)	32.28 (105.9 ft)
Alighting Gear		
Type	Tricycle, fully	Tricycle, fully
	retractable	retractable
Design sink speed, m/sec	2.44 (8 fps)	2.44 (8 fps)
Flight Controls		
Type	Fly-by-wire	Fly-by-wire
	Dual main and	Dual rotor and
	tail rotor servos	fixed wing control servos
	Dual ASE, triply	Dual ASE, triply
	redundant	redundant
Engine Installation		
Type	Rubberized Allison	Rubberized Allison
	501-M62	501-M62
Scale factor	.445	.922
Number	3	3
Location	Two side-mounted,	One behind each fan
	one to rear of	propulsor,
	main gearbox	Third to rear of
		main gearbox
Total installed power, shp	10753 (10605 hp)	22287 (21979 hp)
Output speed, rpm	17244	11978
Fuel System		
Capacity, m³	2.751 (726.7 gals.)	4.230 (1117.4 gals.)
Type	Bladder, pressure	Wet wing, pressure
	refueling, one tank	refueling, one tank
	per engine with	per engine per wing,
	crossfeed. Space	with crossfeed.
	for double volume.	

FIGURE 3-4. BASELINE AIRCRAFT DESCRIPTION

	Helicopter	Compound
Drive System		
Main gearbox	Passive isolation	Passive isolation
	Damped gears	Damped gears
Rating, shp	7425 (7322 hp)	12313 (12143 hp)
Reduction ratio, hover	39.67	30.18
Reduction ratio, cruise	39.67	40.0
Input speed, rpm	6000	4500
Output speed, rpm	151.2	149.1 (hover)
Intermediate gearbox		
Design power, shp	1708	2820
Input speed, rpm	4500	4500
Output speed, rpm	2195	4500
Reduction ratio	2.05	1.0
Tail gearbox		
Design power, shp	1708	2820
Input speed, rpm	2195	4500
Output speed, rpm	731.5	555.1
Reduction ratio	3.0	8.1
Engine reduction gearbox		
Reduction ratio	2.874	2.662
		(Reduction taken
		through fan gear-
		box takeoff for
		wing engines)
Auxiliary Propulsion		
Type	-	Hamilton Standard Q-Fan
Diameter, m	-	2.423 (7.95 ft)
Number	-	2
Blade tip speed, m/sec	-	213.4 (700 fps)
rpm	-	1681.1
Gearbox reduction ratio	-	7.125
APU		Sized to meet needs of starting, ground air conditioning, and heating
Instruments		Full IFR instruments for pilot and co-pilot
Hydraulics		Dual system
Electrical		Dual system
Avionics		VHF/AM radio (2)
		Intercom system
		Public address
		Voice recorder
		VOR/LOC/GS (2)
		ADF
		DME
		ATL transponder
		Radar altimeter (2)
		Gyro compass (2)
		Area nav. system (2)
		Microwave ILS (2)
		Automatic approach coupler
		Marker Beacon
		Flight director (2)
Air Conditioning and Anti-Ice	Environmental Control System as for current fixed wing transport. Windshield and engine inlet anti-ice.	Environmental Control System as for current fixed wing transport. Windshield, engine inlet, wing and empennage leading edge anti-ice.
Furnishings	Passenger seats (100)	Passenger seats (100)
	Crew seats (3)	Crew seats (3)
	Attendant seats (2)	Attendant seats (2)
	Lavatories (2)	Lavatories (2)
	Wall & ceiling trim/ soundproofing	Wall & ceiling trim/ soundproofing
	Floor covering	Floor covering
	Overhead racks	Overhead racks
	Beverage stowage	Beverage stowage
	Partitions	Partitions
	Crew equipment	Crew equipment
	Cabin equipment	Cabin equipment
	Emergency equipment	Emergency equipment
	Baggage accommodation & restraint system	Baggage accommodation & restraint system

FIGURE 3-4. (CONTINUED)

A fuselage section cut, identical for all designs described in this report, is given in Figure 3-5. All designs have canted tail rotor/fan configurations. This feature provides a significant weight saving advantage because of the component of tail lift in hover. Without this, a fuselage extension would be needed forward to maintain the aircraft center of gravity under the main rotor head. Separation between main and tail rotor tip paths is 7 percent of tail rotor diameter, a minimum amount desirable to suppress noise generated from interference between the two flowfields.
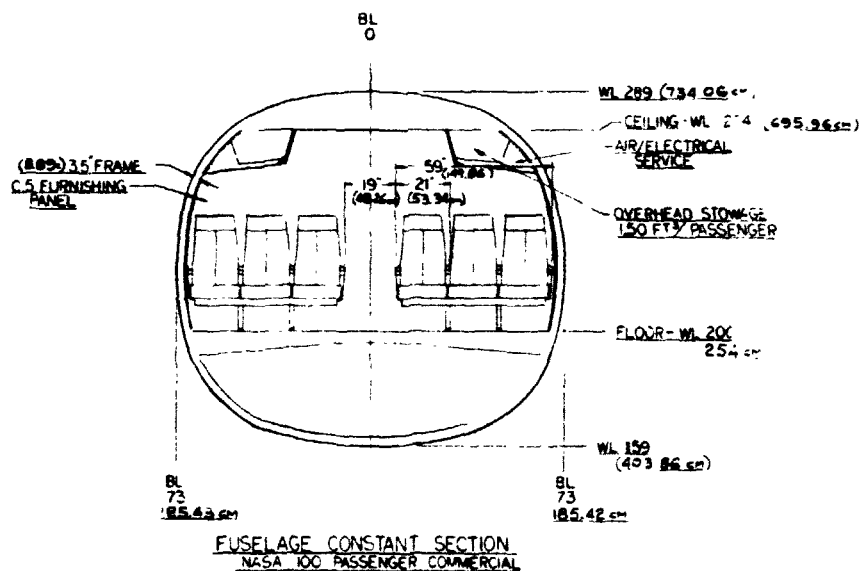

FIGURE 3-5. FUSELAGE SECTION CUT

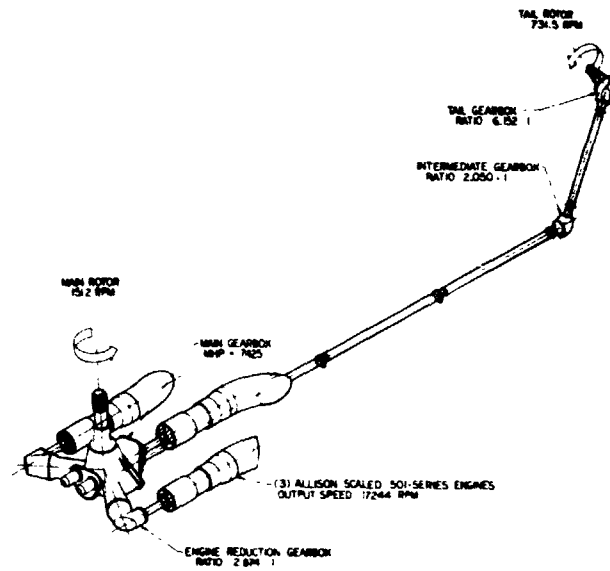Drive system schematics are shown in Figures 3-6 and 3-7.
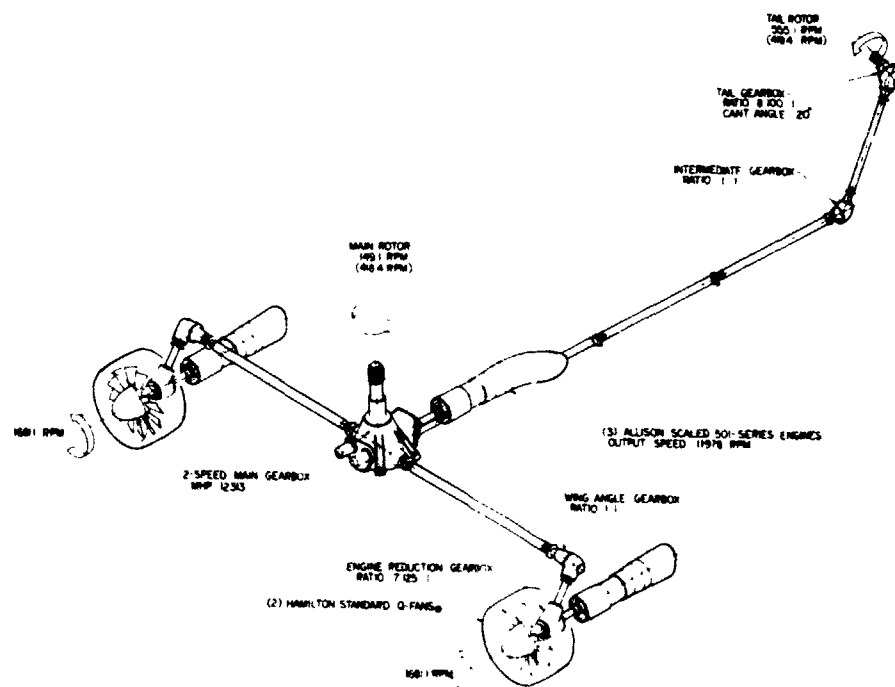

FIGURE 3-6. HELICOPTER DRIVE SYSTEM SCHEMATIC

FIGURE 3-7. COMPOUND DRIVE SYSTEM SCHEMATIC

3.2 Mass Properties

3.2.1 Weight

The weight estimates for both baseline designs, Figure 3-8, are based on statistical or semi-analytical weight equations for all structural and dynamic components. These weight equations have been developed using a large historical data base, which includes over 40 models of 12 basic helicopters. This data base includes aircraft with a gross weight of up to 31,745kg(70,000 lbs), a main rotor with 3 to 7 blades and from 7.3m (24 ft) to 24.1m (79 ft) in diameter, main gearboxes from 253.5 mhp (250 hp) to 11,154 mhp (11,000 hp), structures designed with ultimate load factors up to 5.25g and dive speeds up to 177.5 m/sec (345 knots).

GROUP	HELICOPTER WEIGHT (kg)	WEIGHT (lb)	COMPOUND WEIGHT (kg)	WEIGHT (lb)
MAIN ROTOR GROUP	.313	5099	2434	5367
WING GROUP	0	0	896	1975
TAIL GROUP				
TAIL ROTOR/FAN	168	370	356	785
TAIL SURFACES	377	832	473	1043
BODY GROUP	2988	6587	3666	8082
ALIGHTING GEAR	651	1435	806	1777
FLIGHT CONTROLS	609	1343	827	1824
ENGINE SECTION	237	523	577	1273
PROPULSION GROUP				
ENGINES	916	2020	1526	3365
AIR INDUCTION	33	72	16	36
EXHAUST SYSTEM	20	45	34	74
LUBRICATING SYSTEM	0	0	0	0
FUEL SYSTEM	164	362	112	246
ENGINE CONTROLS	33	72	59	129
STARTING SYSTEM	86	189	155	341
AUXILIARY PROPULSION FANS	0	0	2056	4533
DRIVE SYSTEM	2519	5553	3292	7257
AUXILIARY POWER UNIT	266	586	266	586
INSTRUMENTS	262	577	293	645
HYDRAULICS	70	155	78	173
ELECTRICAL GROUP	330	728	391	862
AVIONICS	298	658	298	658
FURNISHINGS	2505	5523	2747	6055
AIR CONDITIONING AND ANTI-ICE	708	1561	733	1617
AUXILIARY GEAR	20	43	20	43
CONTINGENCY	19	41	371	818
WEIGHT EMPTY	15592	34374	22482	49564
FIXED USEFUL LOAD				
PILOT	86	190	86	190
CO-PILOT	86	190	86	190
OIL-ENGINE	18	40	18	40
-TRAPPED	7	16	7	16
FUEL-TRAPPED	5	12	33	73
ATTENDANTS	127	280	127	280
MISSION EQUIPMENT	136	300	136	300
PAYLOAD	8165	18000	8165	18000
FUEL-USABLE	2149	4735	3300	7273
GROSS WEIGHT	26371	58137	34440	75926

FIGURE 3-8. BASELINE DESIGN WEIGHT STATEMENT

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The auxiliary propulsion fan weights are based on trends provided by Hamilton Standard and are based on the current Q-fan demonstrator program. The furnishings and equipment weights for both baseline designs are estimated from data available on the Sikorsky S-65-200 commercial compound program (Reference 6) and from current fixed-wing commercial transports (References 14 and 15).

3.2.2 Balance & Loadability

The forward and aft center of gravity limits are based on the following criteria:

(a) Steady state main rotor flapping in hover should not exceed ±3.75 degrees at any gross weight.

(b) Pitch attitude in hover should not exceed 6 degrees at any gross weight.

The flapping limits are established by rotor hub and shaft fatigue considerations. Attitude limits are established by pilot comfort and visibility considerations. Application of these criteria result in the center of gravity limits at design gross weight as illustrated in Figure 3-9, shown as a function of rotor shaft incidence (3 degrees forward for the helicopter, 0 degrees for the compound).

The center of gravity envelopes for the helicopter and compound are shown in Figures 3-10 and 3-11. The passenger loading follows the generally accepted pattern in which window seats are filled first, then the aisle seats, and finally the remaining seats. Both forward and rear loading capability has been evaluated along with the most critical combination of passenger, baggage, and fuel loading. In addition, a tolerance has been allowed on the probable passenger plus fuel loading to consider the in-flight movement of passengers and possible shifts in fuel distribution due to changes in aircraft attitude. The allowable center of gravity range at design gross weight exceeds the minimum requirement, that which would result from a payload shift of ±5% of cabin length.

3.2.3 Moments of Inertia

The helicopter moments of inertia are based on existing data available from the CH-53E, an aircraft of similar size and gross weight. Compound moments of inertia are based on S-65-200 studies (Reference 6). All inertias were scaled to the gross weights of the baseline designs.

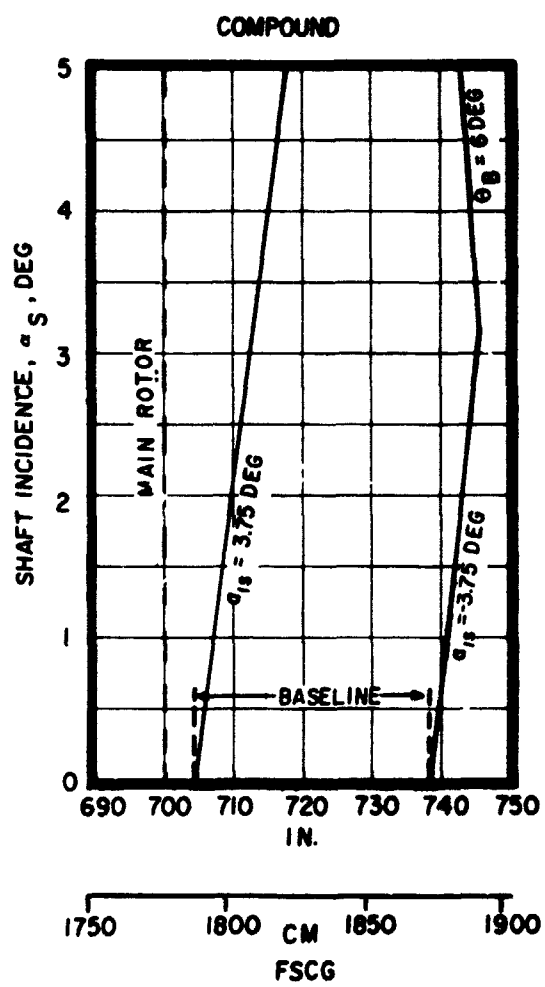|	Inertia - kg cm sec^2 (lb in. sec^2) x 10^6			
	I_{xx} (Roll)	I_{yy} (Pitch)	I_{zz} (Yaw)	I_{xz} (Product)
Helicopter	.936 (.812)	5.29 (4.59)	4.93 (4.28)	.272 (.236)
Compound	2.15 (1.87)	8.47 (7.34)	8.98 (7.79)	.636 (.552)
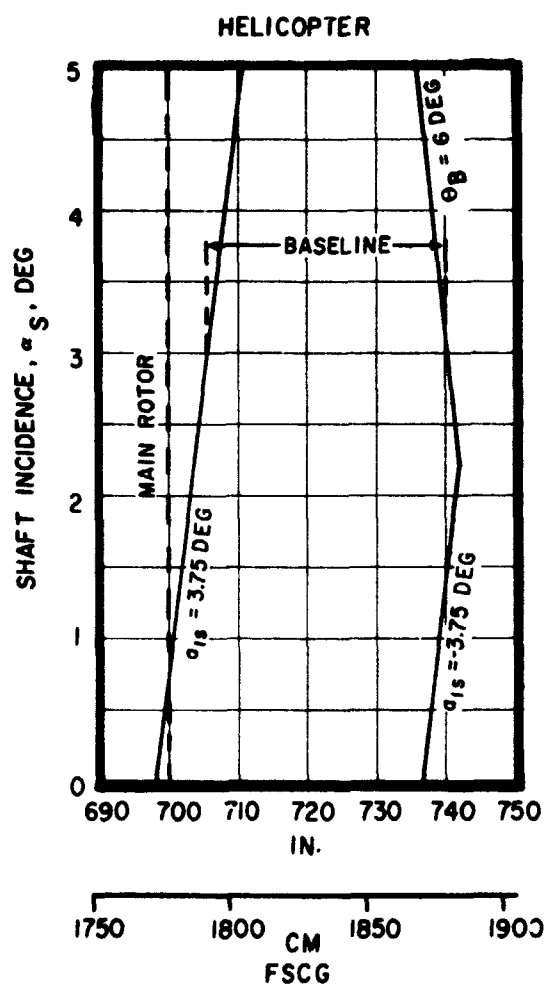
FIGURE 3-9. HOVER C.G. RANGE

25
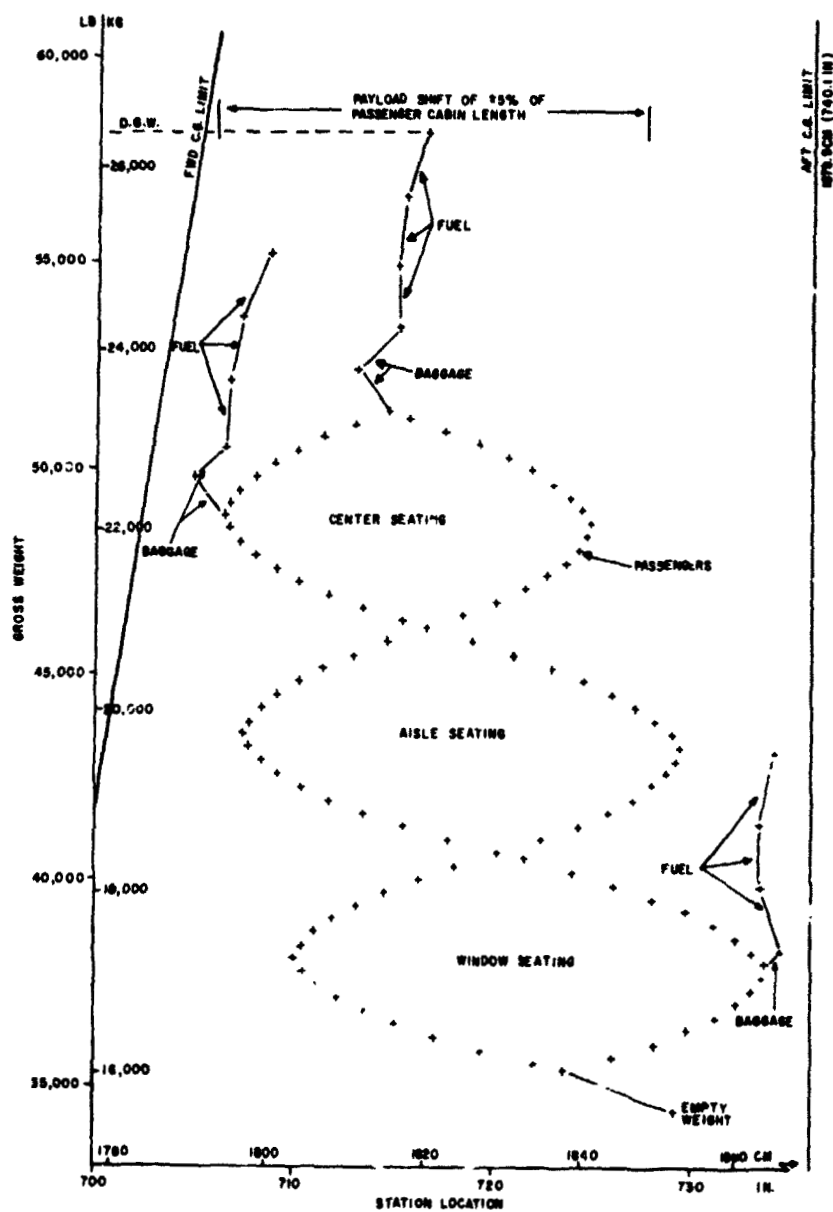
FIGURE 3-10. HELICOPTER BALANCE AND LOADING DIAGRAM

26
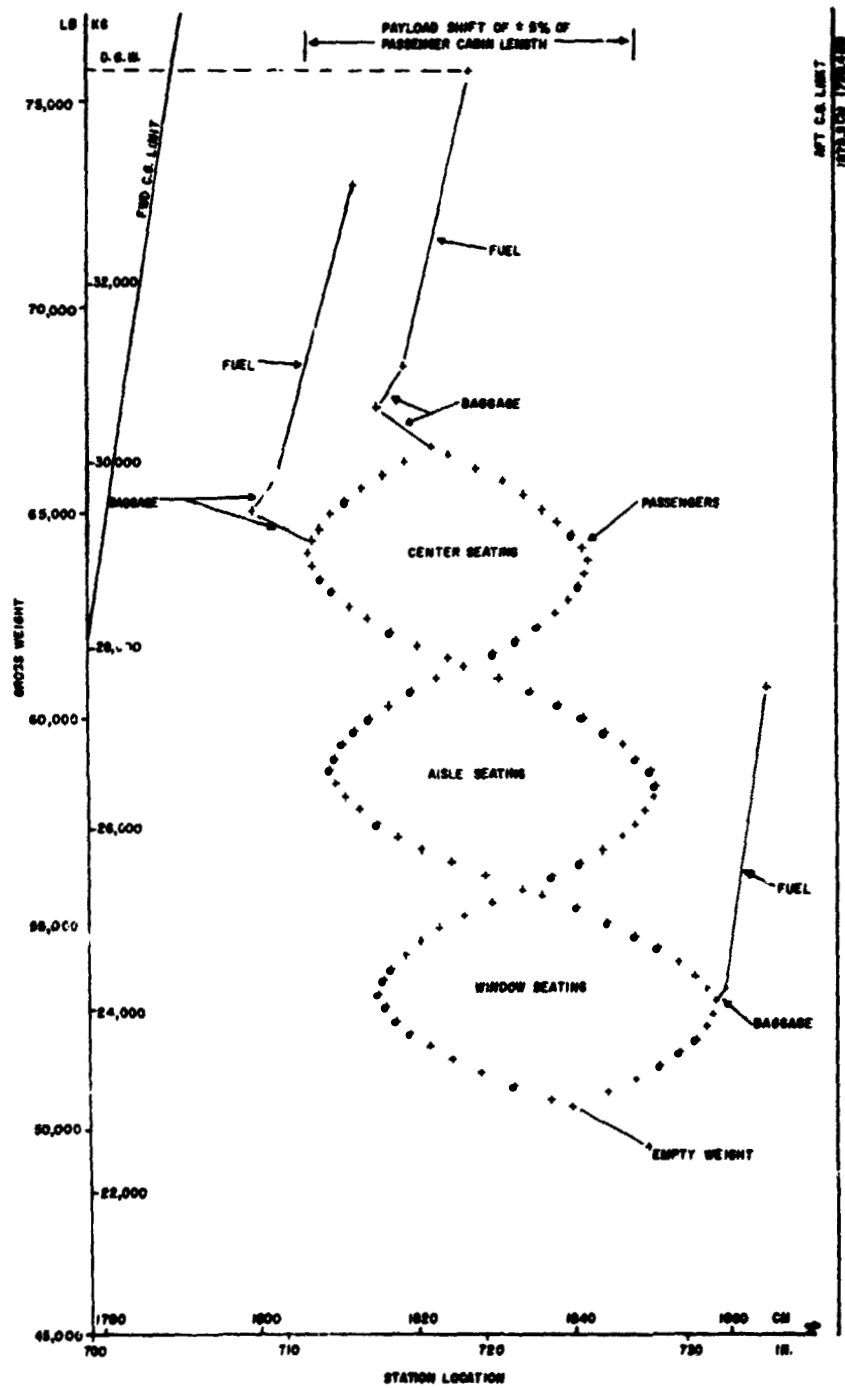
FIGURE 3-11.   COMPOUND BALANCE AND LOADING DIAGRAM

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3.3 Design Optimization and Trending

3.3.1 Helicopter

The helicopter baseline design evolved from a series of design trade-offs generally directed at minimizing DOC. Aspects of design considered to have the most effect on DOC are main rotor disc loading, blade loading, blade twist, and tipspeed, design cruise altitude, tail rotor tipspeed, and number of engines.

3.3.1.1 Disc Loading and Blade Loading

Increasing disc loading reduces rotor diameter and increases hover power requirements. While rotor and drive system weights decrease by reducing the rotor size, this decrease is more than offset by an increased powerplant weight and an increased fuel flow for the larger engine. The resulting trend, Figure 3-12, shows that for any given blade loading, C_T/σ, minimum DOC is always obtained at the lowest possible disc loading. In the study, however, it was found that for disc loadings below 41.5 ksm (8.5 psf), it was not possible to balance the aircraft because of the excess fuselage length aft.

FIGURE 3-12. HELICOPTER - SELECTION OF MAIN ROTOR GEOMETRY

As C_T/σ is increased for any given disc loading, the blade area is reduced, decreasing blade weight. However, reducing the blade area has a serious effect on attainable cruise speed and DOC, as shown in Figure 3-13. A C_T/σ of 0.075 was found to provide the minimum DOC.

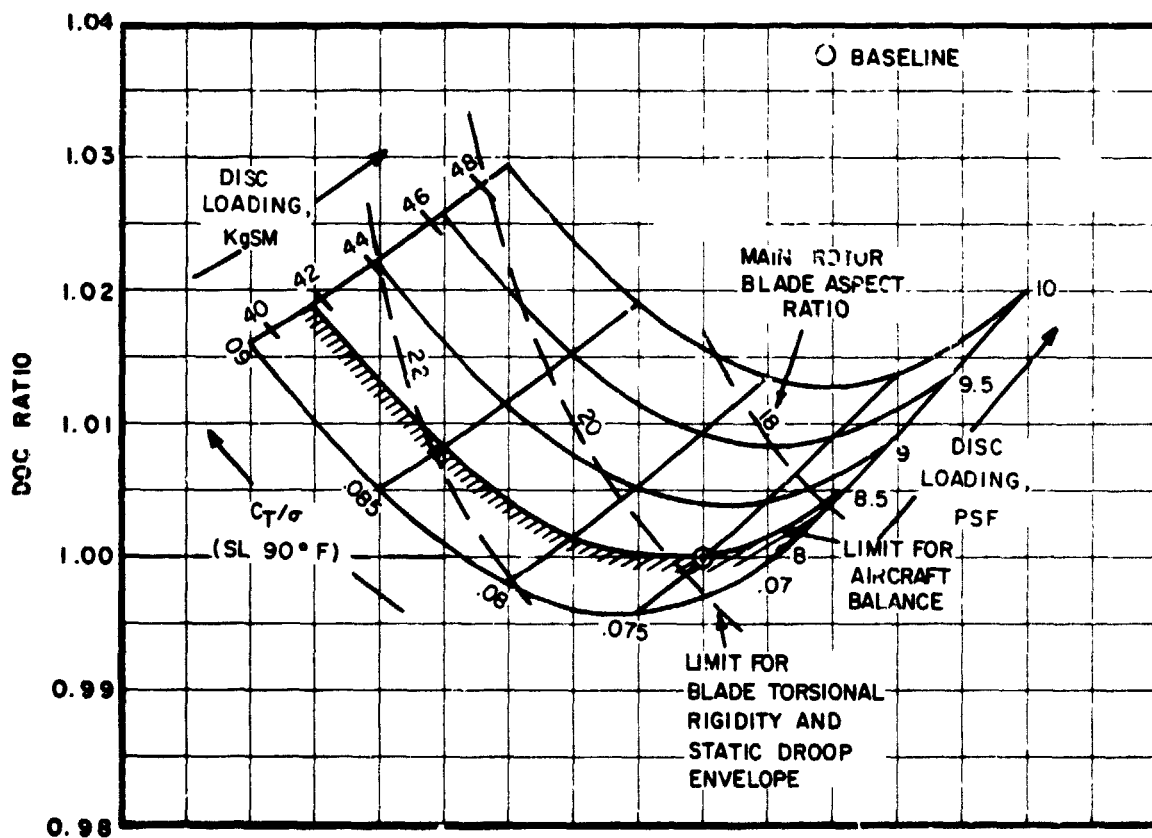Also shown in Figure 3-12 are lines of constant blade aspect ratio. It is seen that at the baseline disc loading and C_T/σ, the blade aspect rat' > is less than the theoretical upper limit of 20, at which blade torsional rig .ity and static droop considerations require substantial increase in blade s .ffness and weight.

3.3.1.2 Cruise Altitude

The trend of DOC with cruise altitude, Figure 3-14, shows a characteristic common to pure helicopters. Because of retreating blade stall, pure helicopters do not exhibit a classic fixed wing increase in cruise efficiency with increasing cruise altitude. For the helicopter, the lowest possible cruise altitude produces the minimum DOC. A cruise altitude of 1219 meters (4000 feet) was selected as the lower limit in order to provide for air traffic control and an altitude safety margin in heavily populated areas.

3.3.1.3 Main Rotor Blade Twist

Blade twist increases rotor efficiency in both cruise and hovering flight. The limit of the DOC versus twist trend, Figure 3-15, occurs when the blade stresses associated with high twist begin to escalate blade weight and when the manufacturing process for such a blade, which generally requires highly non-linear twist and non-linear planform shape, have not been developed. For this study, -16 degrees of twist was selected as that technology level which is currently being developed in the CH-53E and YUH-60A UTTAS rotors and would therefore be available to a large production helicopter in service in 1985.

3.3.1.4 Main Rotor Tipspeed

Selection of main rotor tipspeed is a trade-off between blade weight (for given C_T/σ) and advancing tip Mach number effects in cruise flight. Correct choice of tipspeed is dependent on blade twist, tip shape, and airfoil sections available within the time frame to alleviate the compressibility losses. The optimum tipspeed was 222.5 meters/sec (730 ft/sec), as shown in Figure 3-16.

3.3.1.5 Tail Rotor Tipspeed

DOC does not increase at high tail rotor tipspeeds, (Figure 3-17), because at cruise speeds the tail rotor is not highly loaded and so does not contribute significantly to any power penalty associated with compressibility effects in cruise. Selection of 213 m/sec (700 fps) as an upper limit for tail rotor tipspeed was made in order to suppress tail rotor noise.

3.3.1.6 Number of Engines

As the number of engines increases, the total installed power to provide
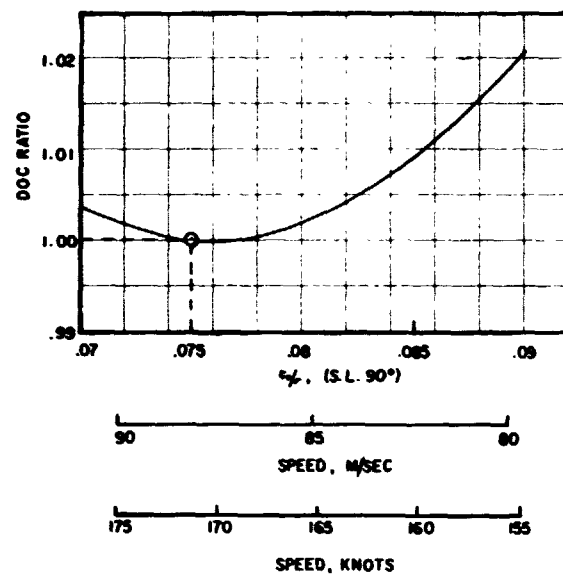
FIGURE 3-13. HELICOPTER - SELECTION OF CRUISE SPEED
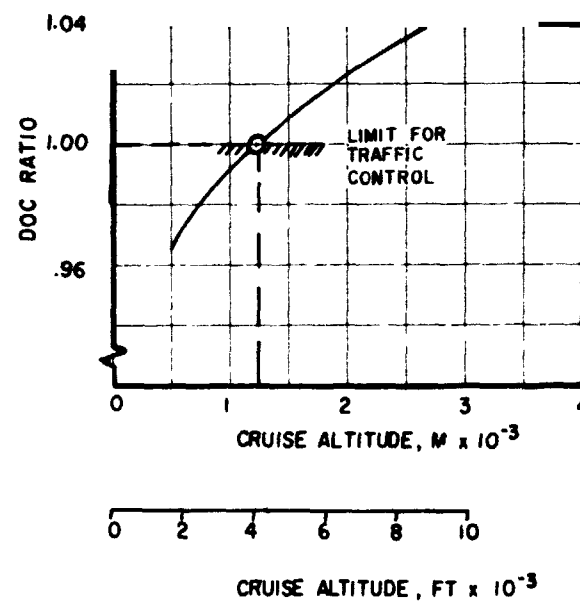

FIGURE 3-14. HELICOPTER - SELECTION OF CRUISE ALTITUDE
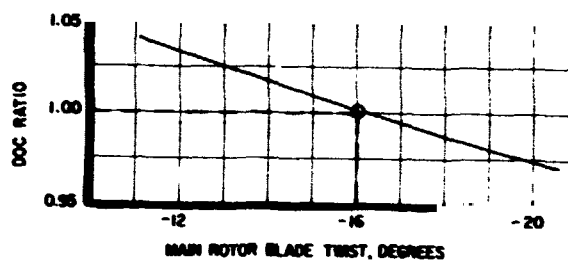
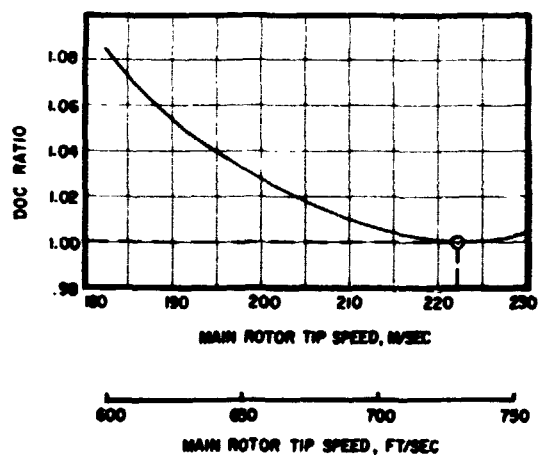FIGURE 3-15.   HELICOPTER - SELECTION OF MAIN ROTOR BLADE TWIST

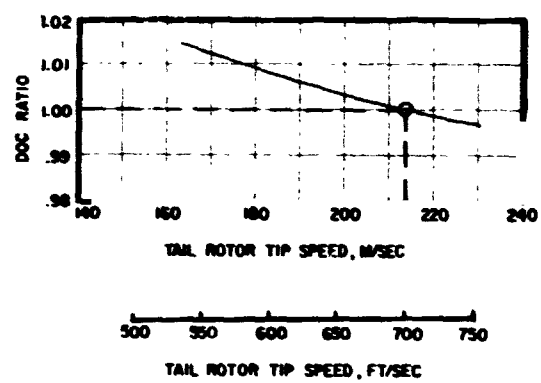FIGURE 3-16.   HELICOPTER - SELECTION OF MAIN ROTOR TIPSPEED

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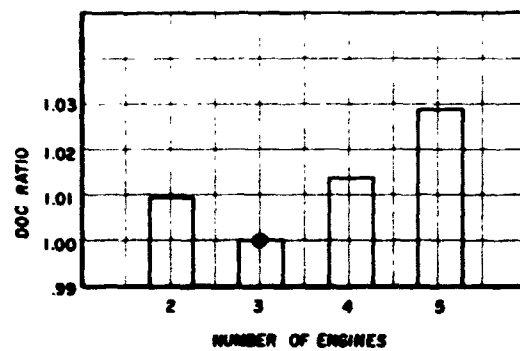FIGURE 3-17. HELICOPTER - SELECTION OF TAIL ROTOR TIPSPEED

FIGURE 3-18. HELICOPTER - SELECTION OF NUMBER OF ENGINES

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the one-engine-inoperative (OEI) out-of-ground-effect (OGE) capability decreases. However, maintenance burden increases. A three-engine solution minimizes DOC, as shown in Figure 3-18.

3.3.2 Compound

The major design parameters investigated were cruise speed, disc loading, prop-fan size and tipspeed, main rotor twist, number of engines, and main rotor tipspeed.

3.3.2.1 Cruise Speed and Disc Loading

For the compound aircraft, the design cruise speed was not selected with minimum DOC alone in mind. This situation is similar to that of a CTOL aircraft, where design cruise speeds are significantly higher than required for minimum DOC, in order to provide a marketable product with superior passenger appeal.

Because main rotor disc loading influences hover efficiency, DOC was trended as a function of cruise speed and disc loading in order to investigate the condition for a power-required match between cruise flight and OEI OGE hover. This is shown in Figure 3-19. Minimum DOC for a given disc loading occurs at the cruise speed for this power match. As a further guide to proper selection of cruise speed, a productivity function of the form (seat kilometers per block hour)/(weight empty) was calculated.

Figure 3-20 shows that this productivity function maximizes at a cruise speed of 128.6 m/sec (250 knots), for 2% increase from the theoretical minimum DOC.

3.3.2.2 Auxiliary Propulsion

Both fan and propeller propulsion were trended in determining the lowest DOC design. A family of Hamilton Standard Q-fan devices, References 12 and 13, was investigated, in terms of the fan pressure ratio, Figure 3-21. A fan pressure ratio of 1.1 yielded the lowest DOC consistent with a shroud diameter that did not compromise the overall vehicle design. The selected fan diameter of 2.4 meters (7.32 ft) was used as the basis for comparison with the propeller configuration.

As discussed further in Section 3.5.2, one of the most important influences on the selection of an auxiliary propulsion device is the requirement for cabin noise suppression to 70 dB PSIL. Figure 3-22 shows the powerful effect on required soundproofing of the tipspeed of the propulsive device. In evaluating the propellers, the required soundproofing weight increases rapidly, when compared to the fans, as tipspeed is increased. Propeller efficiency generally decreases with decreasing tipspeed, but can be restored when activity factor is increased by adding blades or blade chord, and so system weight. Another important noise effect is the proximity of the propeller to the fuselage. The closer the propeller is to the fuselage, the greater the weight penalty for acoustic insulation. For the fans, the acoustic treatment of the shroud, 45.4
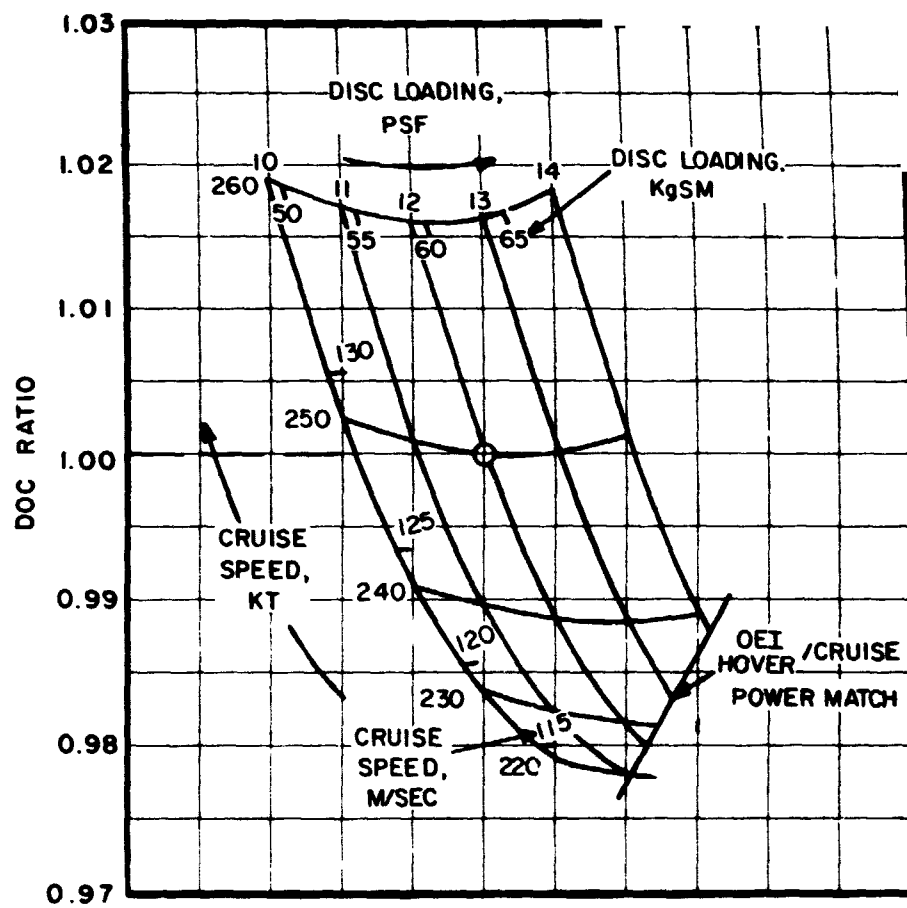
FIGURE 3-19. COMPOUND - SELECTION OF DISC LOADING
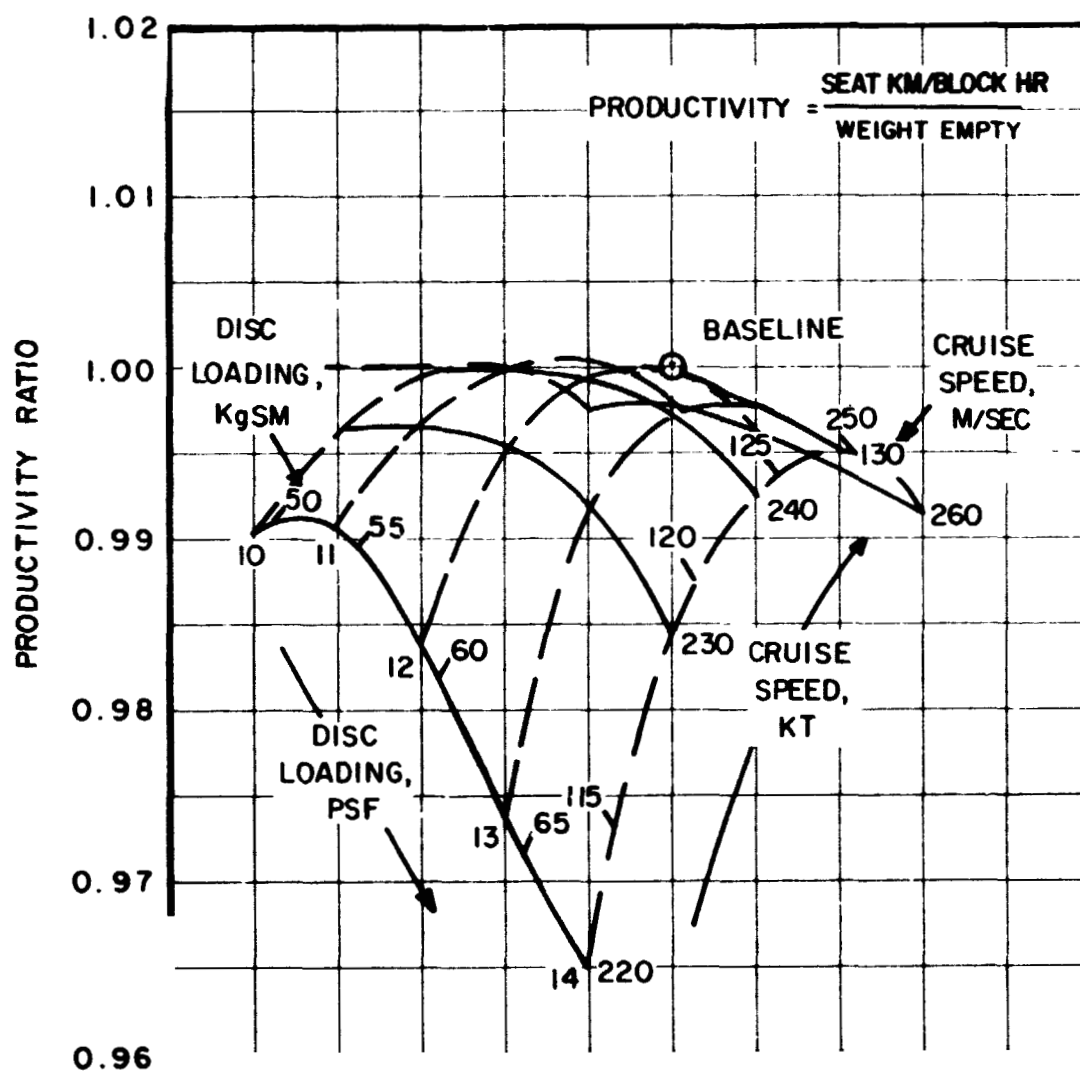
FIGURE 3-20. COMPOUND - SELECTION OF CRUISE SPEED

1.08
1.06
1.04
1.02
1.00
.98

DOC RATIO

1.00
1.10
1.20

FAN PRESSURE RATIO

3 2.8 2.6 2.4   2.2   2.1   2
10   9   8   7.5   7   6.5

FAN DIAMETER, FT : M

.78   .75   .7   .65

PROPULSIVE EFFICIENCY
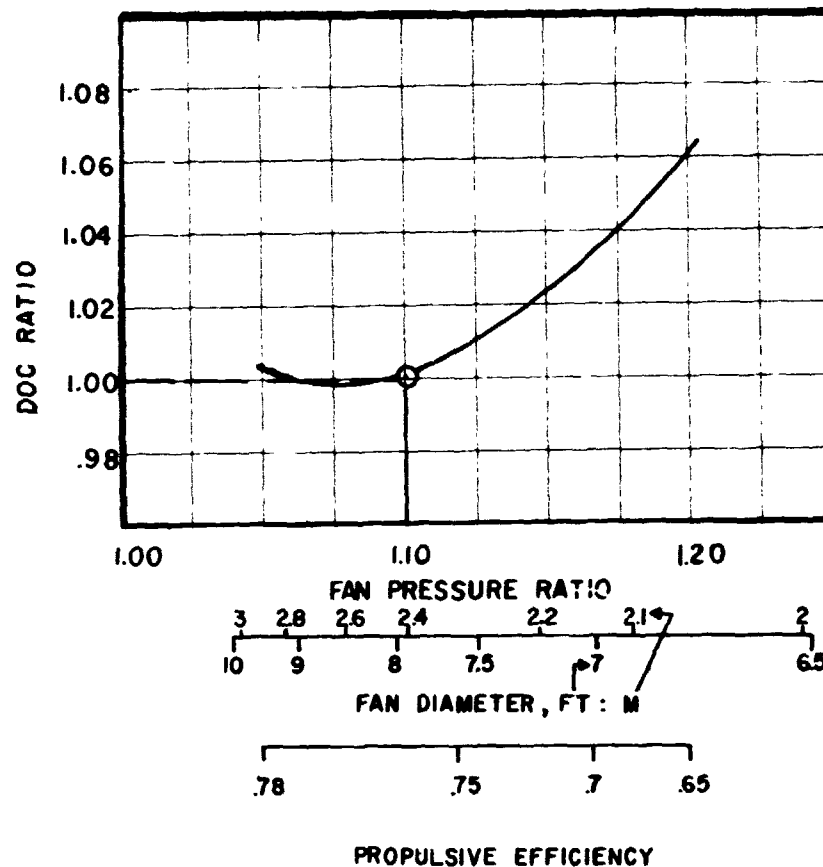
FIGURE 3-21. COMPOUND - SELECTION OF FAN AUXILIARY PROPULSION

kilograms (100 pounds) per propulsor, has been added to the cabin treatment
weight to provide an equivalent average treatment density. Also shown in
Figure 3-22 are the soundproofing treatment densities required to meet the
75 dB PSIL limit during take-off, for various levels of transmission acoustic
isolation. Unlike the helicopter, power to the main gearbox during cruise is
low, and so the contribution of the internal noise signature from the trans-
mission is significant only during low-speed flight. While this dictates the
soundproofing required for the helicopter, the fan or propeller noise dictates
the soundproofing weight for a compound.

     With these noise penalties assigned, the ratio of DOC to the baseline fan
solution was trended for varying propeller tipspeeds and nacelle buttlines,
Figure 3-23. The extremely high DOCs are associated with a high tipspeed in
combination with close proximity of propeller and fuselage. The correspondence
between propeller diameter and nacelle buttline is provided by propeller/ground
and propeller/rotor tip path clearance considerations.

     With the noise constraints, the best propeller solution is marginally
competitive to the prop-fan design, which provides reduced system envelope,

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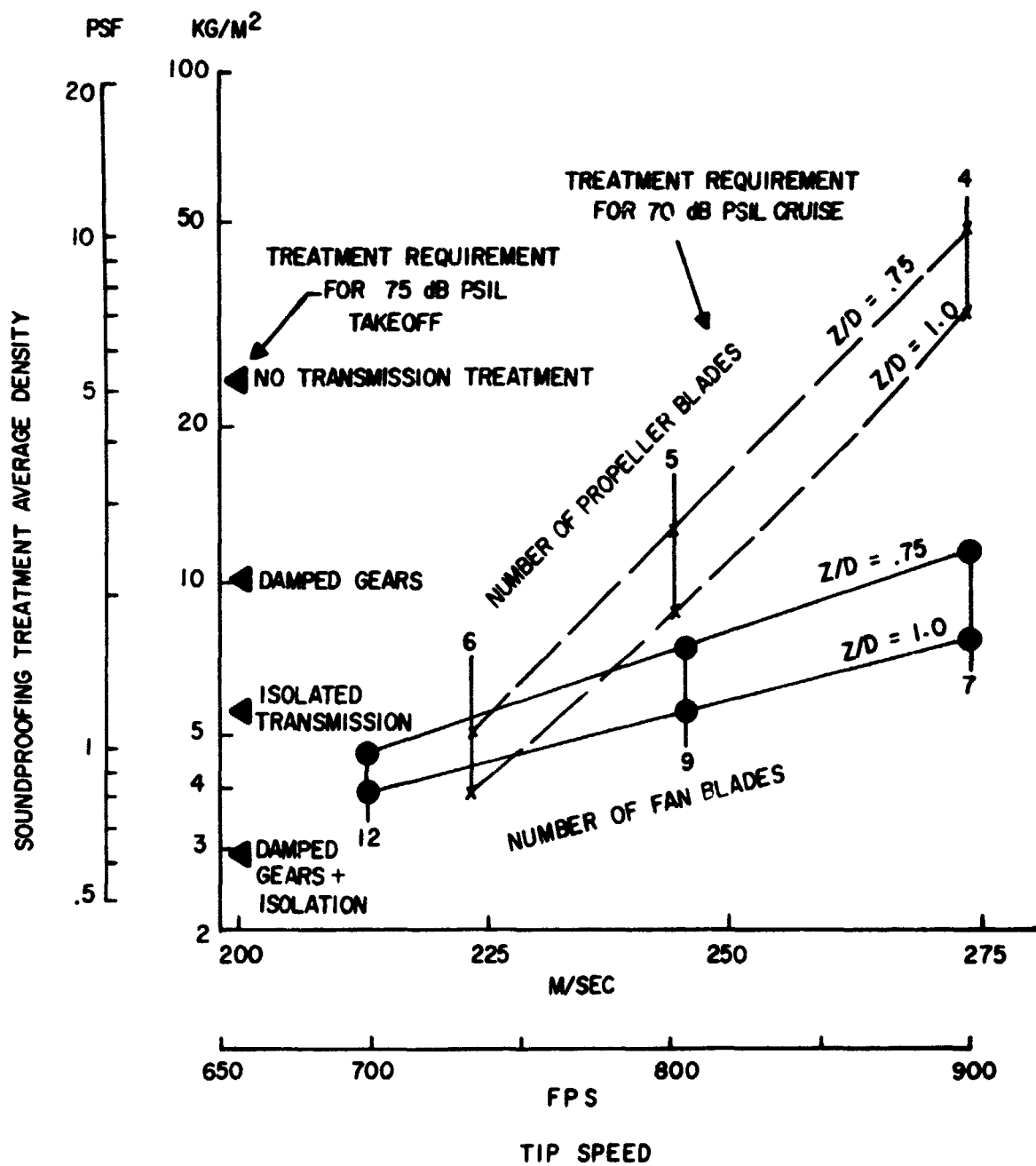
FIGURE 3-22. COMPOUND - INTERNAL ACOUSTIC TREATMENT

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reduced slipstream interference effects, and increased clearance from the ground
and rotor tip-path. These factors weigh heavily in favor of the prop-fan, in
spite of its reduced propulsive efficiency at the compound cruise speed.
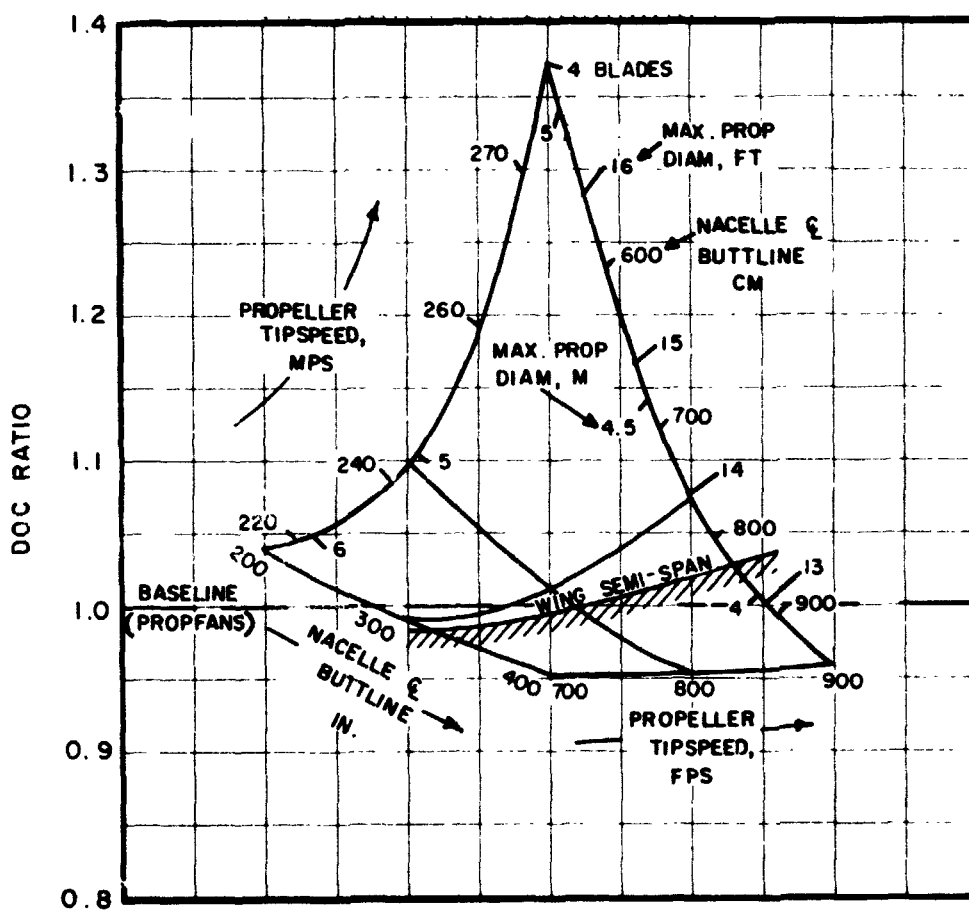
FIGURE 3-23 COMPOUND - PROPELLER/FAN TRADE-OFF

3.3.2.3 Main Rotor Blade Twist

In the design of a compound rotor with constant geometry, high blade twist
is not mandatory, because the higher design cruise power requirements of a
compound allow the inefficiencies in hover of lower twist designs without
increasing rotor size. Twist is also undesirable because of the high stresses
in a twisted blade at cruise speeds and high rotor inplane drag forces. The

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minimum DOC occurs at a twist of -4 degrees, Figure 3-24.

While from a performance point of view, -4 degrees of twist is optimum, the external noise constraints of this study required analysis of all design parameters to identify ways to lower noise. Analysis of twist showed that higher values of twist reduced external noise. This is due to a more even aerodynamic distribution of lift along the blade radius, which reduces the tip vortex strength. Twist of -12 degrees was selected because of its powerful effect in reducing noise, while the flatness of the DOC trend in Figure 3-24 indicates only a 1% increase in DOC above the minimum.

3.3.2.4 Number of Engines

A three-engine installation was selected for the compound, Figure 3-25. One engine is centrally installed behind the main gearbox, and one is axially mounted behind each auxiliary propulsor. Four-engine and five-engine solutions were studied but rejected, because their maintenance burden drove DOCs to a relatively high level compared with the two-engine and three-engine solutions. The two-engine solution was not selected. Although it yielded a slightly lower DOC, the physical size of an engine having one-half of the required installed horsepower was prohibitive. The enormous size created installation problems and fan losses, which were in violation of the performance and weights assumptions of the analysis. In addition to this, the size of the engine violated the study groundrule of development by 1985. The engines were more than 50 percent larger than the baseline (HLH development) engines.

3.3.2.5 Main Rotor Tipspeed

The compound main gearbox has a two-speed input section. This provides a higher rotor tipspeed for hover and helicopter flight up to 92.6 m/sec (180 knots) and a lower rotor tipspeed in high-speed flight in order to reduce advancing tip Mach number effects. Figure 3-26 shows the trend of DOC with hover tipspeed, indicating a minimum at 222.5 m/sec (730 fps). As discussed in Section 3.5, the compound design must be compromised to reduce external noise signature to the 95 PNdB limit. The baseline tipspeed is thus 210.3 m/sec (690 fps).

3.3.2.6 Tail Rotor Tipspeed

The variation of DOC with tail rotor tipspeed, Figure 3-27, shows a minimum at 210.3 m/sec (690 fps). This tipspeed does not violate the requirements for limiting external noise to 95 PNdB.

3.3.2.7 Main Rotor Blade Loading

The main rotor blade loading, defined as the C_T/σ value in hover on a sea level 90-degree day, has a powerful influence on external noise signature (Section 3.5). Unlike the helicopter, the compound rotor is unloaded in high speed flight so that blade stall is not a limiting criteria. In fact, excess blade area merely produces inplane rotor drag. Also there is an excess of power available in hover, because the installed power is determined by the high-
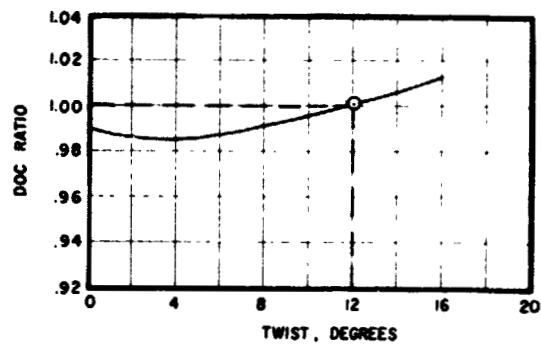
FIGURE 3-24.   COMPOUND - SELECTION OF MAIN ROTOR BLADE TWIST
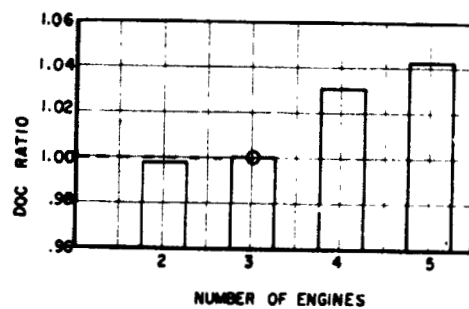

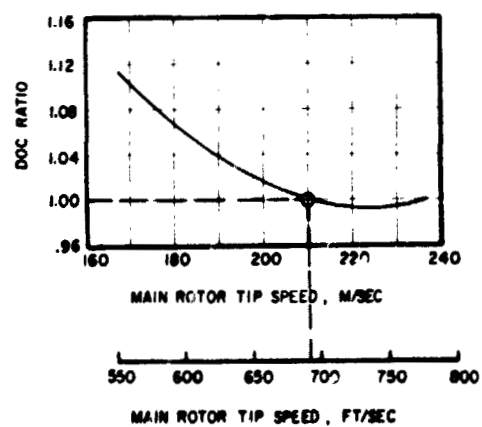FIGURE 3-25.   COMPOUND - SELECTION OF NUMBER OF ENGINES

FIGURE 3-26.   COMPOUND - SELECTION OF MAIN ROTOR TIPSPEED
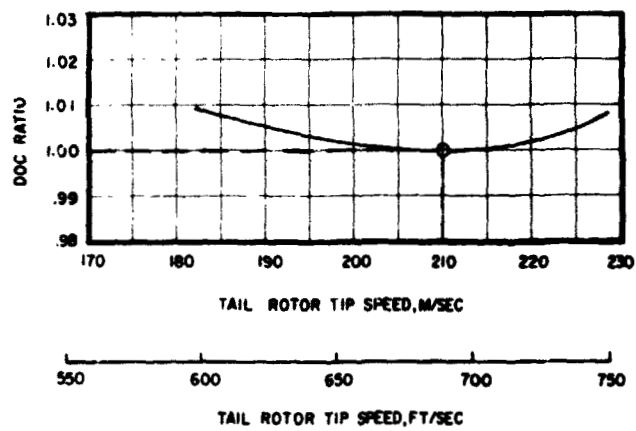
FIGURE 3-27. COMPOUND - SELECTION OF TAIL ROTOR TIPSPEED
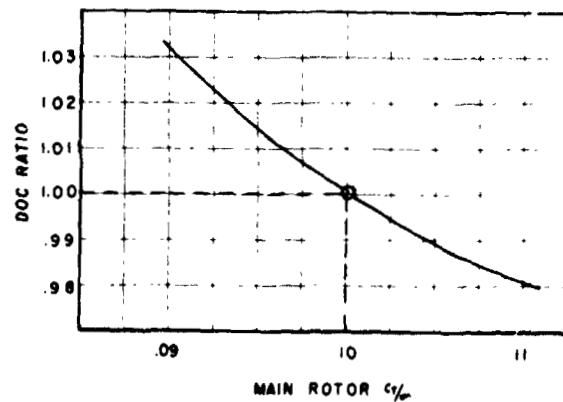
FIGURE 3-28. COMPOUND - SELECTION OF HOVER BLADE LOADING

speed requirements. Hover inefficiencies (high C_T/σ) can therefore be tolerated in the interests of improving the high speed capability of the aircraft and reducing weight. This is indicated in Figure 3-28. DOC continues to decrease as C_T/σ increases beyond a value of .115. Because of the external noise limit, however, hover C_T/σ was limited to 0.1.

3.4 Handling Qualities and Gust Sensitivity

3.4.1 Handling Qualities Criteria

The helicopter and compound helicopter were designed to meet the level 1 requirements of Appendix A of Reference 2 at all times. A cross reference between these requirements and this text is given in Figure 3-29. Each aircraft is designed to continue flight at normal rotor rpm with one engine inoperative, thus not degrading roll and pitch control powers. If the power required for a maneuver exceeds power available for a short time, the rotational energy stored in the rotor systems can be used for yaw and vertical height control.

Each aircraft i. designed with a triply-redundant automatic flight control system with voting. Therefore, any failure is automatically detected and the appropriate element of the system is shut off without degrading control system performance.

Level one is defined for normal operation by an average commercial pilot. Level two requires adequate flying qualities with increased pilot work load, after any reasonable failure of a single gas generator or control system element.

TEXT	REFERENCE 2
3.4.1	1.1
3.4.2	1.1.1
3.4.3.1	1.1.2.1
3.4.3.2	1.1.2.2
3.4.4	1.1.3
3.4.5	1.1.4
3.4.6	1.1.5
3.4.7	1.2
3.4.8	4.8
3.4.9	5.1
3.4.10	5.2
3.4.11	5.4

FIGURE 3-29. CROSS REFERENCE TO STUDY GUIDELINES HANDLING QUALITIES REQUIREMENTS

3.4.2 Attitude Control Power

Trim data, in combination with the control limits, Figure 3-30, and the control systems derivatives, Figure 3-31, were used to calculate the maximum angular acceleration about each axis for the control available in hover, 80 knots, and 171 knots for the helicopter; and in hover, 100 knots, and 180 knots for the compound, at their forward and aft centers of gravity, Figures 3-32 and 3-33. As can be seen, the angular acceleration available exceeds that required in all cases with the control ranges selected for this study. With a rotary wing aircraft, it is possible to apply 100% of each of the controls simultaneously.

The control ranges used in this study are based on the range required for the Sikorsky CH-53E (an aircraft of similar size). This control system will provide the aircraft with satisfactory control harmony and sensitivity, as required by Reference 10.

CONTROL LIMITS	
Helicopter	
A_{1s}	± 8 DEG.
B_{1s}	-12 to +16 DEG.
$\Theta TR_{.75R}$	-10 to +30 DEG.
Compound	
A_{1s}	± 8 DEG.
B_{1s}	-11 to +18.5 DEG.
$\Theta TR_{.75R}$	-10 to +30 DEG.

FIGURE 3-30. CONTROL LIMITS

CONTROL DERIVATIVES			
HELICOPTER			
VELOCITY M/SEC (kts)	$\dfrac{\partial L/\partial A_{1s}}{I_{xx}\ (57.3)}$	$\dfrac{\partial M/\partial B_{1s}}{I_{yy}\ (57.3)}$	$\dfrac{\partial N/\partial \Theta_{TR}}{I_{zz}\ (57.3)}$
0 (0)	.357	-.0624	-.0466
41 (80)	.340	-.0764	-.0592
88 (171)	.323	-.1215	-.0797
COMPOUND			
VELOCITY M/SEC (kts)	$\dfrac{\partial L/\partial A_{1s}}{I_{xx}\ (57.3)}$	$\dfrac{\partial M/\partial B_{1s}}{I_{yy}\ (57.3)}$	$\dfrac{\partial N/\partial \Theta_{TR}}{I_{zz}\ (57.3)}$
0 (0)	.1725	-.0429	-.0463
51.4 (100)	.1378	-.08616	-.0577
92.6 (180)	.1085	-.15226	-.0774

FIGURE 3-31. CONTROL DERIVATIVES

SPEED M/SEC (KTS)	C.G.	AXIS	REQUIRED ACCELERATION RAD/SEC^2	AVAILABLE ACCELERATION RAD/SEC^2
0 (0)	AFT	ROLL	$\pm.6$	3.32 -2.39
		PITCH	$\pm.33$	0.94 -0.81
		YAW	$\pm.25$	1.10 -0.77
	FWD	ROLL	$\pm.6$	3.57 -2.14
		PITCH	$\pm.33$	0.56 -1.18
		YAW	$\pm.25$	1.10 -0.77
41 (80)	AFT	ROLL	$\pm.4$	3.06 -2.38
		PITCH	$\pm.3$	1.43 -0.71
		YAW	$\pm.2$	0.85 -1.52
	FWD	ROLL	$\pm.4$	3.23 -2.21
		PITCH	$\pm.3$	1.06 -1.08
		YAW	$\pm.2$	C.88 -1.49
88 (171)	AFT	ROLL	$\pm.4$	2.94 -2.23
		PITCH	$\pm.3$	3.05 -0.35
		YAW	$\pm.2$	0.90 -2.29
	FWD	ROLL	$\pm.4$	3.07 -2.10
		PITCH	$\pm.3$	2.70 -0.70
		YAW	$\pm.2$	0.64 -2.23

FIGURE 3-32. BASELINE HELICOPTER CONTROL POWER

4,

SPEED M/SEC (KTS)	C.G.	AXIS	REQUIRED ACCELERATION RAD/SEC^2	AVAILABLE ACCELERATION RAD/SEC^2
0 (0)	AFT	ROLL	$\pm.6$	1.66 -1.10
		PITCH	$\pm.33$	0.64 -0.62
		YAW	$\pm.25$	0.93 -0.92
0 (0)	FWD	ROLL	$\pm.6$	1.78 -0.98
		PITCH	$\pm.33$	0.39 -0.88
		YAW	$\pm.25$	0.92 -0.94
51.4 (100)	AFT	ROLL	$\pm.4$	1.38 -0.83
		PITCH	$\pm.3$	1.80 -0.74
		YAW	$\pm.2$	0.61 -1.70
	FWD	ROLL	$\pm.4$	1.46 -0.74
		PITCH	$\pm.3$	1.46 -1.08
		YAW	$\pm.2$	0.61 -1.70
92.6 (180)	AFT	ROLL	$\pm.4$	0.99 -0.75
		PITCH	$\pm.3$	3.14 -1.36
		YAW	$\pm.2$	1.02 -2.08
	FWD	ROLL	$\pm.4$	1.08 -0.65
		PITCH	$\pm.3$	2.73 -1.77
		YAW	$\pm.2$	1.02 -2.08

FIGURE 3-33. BASELINE COMPOUND CONTROL POWER

From time histories o. .esponses to control stick inputs at the rotor head, displacements after one second are shown in Figures 3-34 and 3-35.

BASELINE HELICOPTER MAXIMUM ANGULAR DISPLACEMENT - 1 SECOND AFTER A CONTROL STEP INPUT SAS OFF				
SPEED M/SEC (KT)	CENTER OF GRAVITY POSITION	AXIS	REQUIREMENT DEG	AVAILABLE DEG
0 (0)	AFT	PITCH	±3	22.5 -19.5
		ROLL	±5	55.8 -40.2
		YAW	±2	31.2 -22.0
	FWD	PITCH	±3	13.5 -28.5
		ROLL	±5	60.0 -36.0
		YAW	±2	31.3 -22.0
88 (171)	AFT	PITCH	±3	50.2 - 5.8
		ROLL	±3	54.6 -41.4
		YAW	±2	17.0 -43.0
	FWD	PITCH	±3	44.4 -11.6
		ROLL	±3	57.0 -39.0
		YAW	±2	12.0 -42.0

FIGURE 3-34. BASELINE HELICOPTER MAXIMUM ANGULAR DISPLACEMENT ONE SECOND AFTER A CONTROL STEP INPUT

BASELINE COMPOUND MAXIMUM ANGULAR DISPLACEMENT – 1 SECOND AFTER A CONTROL STEP INPUT SAS OFF				
SPEED M/SEC (KT)	CENTER OF GRAVITY POSITION	AXIS	REQUIREMENT DEG	AVAILABLE DEG
0 (0)	AFT	PITCH	±3	16.0 -15.5
		ROLL	±5	35.6 -23.7
		YAW	±2	25.5 -25.2
	FWD	PITCH	±3	9.6 -21.9
		ROLL	±5	38.1 -21.1
		YAW	±2	25.1 -25.6
92.6 (180)	AFT	PITCH	±3	36.1 -15.6
		ROLL	±3	20.0 -15.2
		YAW	±2	18.3 -37.7
	FWD	PITCH	±3	31.3 -2 .3
		ROLL	±3	22.0 -13.2
		YAW	±2	18.3 -37.7

FIGURE 3-35.   BASELINE COMPOUND MAXIMUM ANGULAR DISPLACEMENT ONE SECOND AFTER A CONTROL STEP INPUT

Figures 3-36 and 3-37 give available control accelerations in a ±12.87 m/sec (25-knot) sidewind, showing control power capabilities substantially in excess of the requirements.

SPEED M/SEC (KT)	V_y M/SEC (KT)	AXIS	REQUIRED ACCELERATION RAD/SEC^2	AVAILABLE CONTROL DEG.	AVAILABLE ACCELERATION RAD/SEC^2
0 (0)	12.87 (25)	ROLL	±.30	7.7	2.7'
				- 8.3	-2.96
		PITCH	±.165	12.6	-0.78
				-15.4	0.96
		YAW	±.125	12.9	-0.60
				-27.1	1.26
0 (0)	-12.87 (-25)	ROLL	±.3	10.3	3.68
				- 5.7	-2.03
		PITCH	±.1165	12.4	-0.77
				-15.6	0.97
		YAW	±.125	23.4	-1.09
				-16.6	0.77
41 (80)	12.87 (25)	ROLL	±.2	7.6	2.58
				- 8.4	-2.86
		PITCH	±.15	9.6	-0.73
				-18.4	1.40
		YAW	±.1	23.3	-1.38
				-16.7	0.98
41 (80)	-12.87 (-25)	ROLL	±.2	10.2	3.47
				- 5.2	-1.77
		PITCH	±.15	9.6	-0.73
				-18.4	1.40
		YAW	±.1	32.3	-1.91
				- 7.7	0.46
88 (171)	12.87 (25)	ROLL	±.2	6.5	2.10
				- 9.5	-3.07
		PITCH	±.15	3.4	-0.41
				-24.6	2.99
		YAW	±.1	27.8	-2.21
				-12.7	1.01
88 (171)	-12.87 (-25)	ROLL	±.2	12.65	4.08
				- 3.35	-1.08
		PITCH	±.15	3.9	-0.47
				-24.1	2.93
		YAW	±.1	32.8	-2.61
				- 7.2	0.57

FIGURE 3-36. BASELINE HELICOPTER CONTROL POWER IN ±25-KT CROSSWIND

SPEED M/SEC (KT)	V_Y M/SEC(KT)	AXIS	REQUIRED ACCELERATION RAD/SEC^2	AVAILABLE ACCELERATION RAD/SEC^2
0 (0)	12.87(25)	ROLL	±.3	1.29 -1.47
		PITCH	±.165	-0.54 0.72
		YAW	±.125	-0.77 1.08
0 (0)	-12.87(-25)	ROLL	±.3	1.90 -0.86
		PITCH	±.165	-0.55 0.71
		YAW	±.125	-1.26 0.59
51.4 (100)	12.87(25)	ROLL	±.2	1.09 -1.12
		PITCH	±.15	-0.69 1.85
		YAW	±.1	-1.61 -0.70
51.4 (100)	12.87(-25)	ROLL	±.2	1.59 -0.62
		PITCH	±.15	-0.78 1.77
		YAW	±.1	-1.97 0.33
92.6 (180)	12.87(25)	ROLL	±.2	0.77 -0.96
		PITCH	±.15	-1.61 3.18
		YAW	±.1	-2.55 0.55
92.6 (180)	12.87(-25)	ROLL	±.2	1.28 -0.46
		PITCH	±.15	-1.48 3.01
		YAW	±.1	-2.81 0.29

FIGURE 3-37. BASELINE COMPOUND CONTROL POWER IN ±25-KT CROSSWIND

3.4.3 Low Speed Control

3.4.3.1 Flight Path Control Power

In the hover to 40 knots speed range, vertical flight control is obtained with collective pitch independently of attitude control power. To obtain an incremental acceleration for height control of ±.1g, 0.75 degrees of collective pitch is required for the helicopter and 0.98 degrees for the compound. The helicopter and compound have ample control margins remaining to obtain this acceleration. The helicopter has +7.2 to -7.8 degrees of collective pitch remaining, and the compound has +4.8 to -11.2 degrees.

With wheels just clear of the ground, the collective pitch required for trim will be slightly lower than in free air, due to ground effect, but more than enough collective pitch remains to produce an incremental acceleration of -.10g to +.05g. In fact, the safety of both configurations is greatly enhanced by the fact that they can both achieve +1.35g's vertically with the wheels just clear of the gound.

The longitudinal cyclic control system was designed to provide at least the longitudinal incremental acceleration of ±.15g required by Reference 2. The capabilities of the helicopter and compound are shown in Figure 3-38 for hover and 40 knots trim flight condition. These capabilities exceed or equal the requirements of Reference 2 and are independent of the loss of an engine in the low speed range.

Velocity M/SEC (Knots)	C.G.	Longitudinal Acceleration Required	Longitudinal Acceleration Available
HELICOPTER			
0 (0)	Aft	±.15	.227 -.262
	Fwd	±.15	.332 -.157
20.6 (40)	Aft	±.15	.182 -.308
	Fwd	±.15	.294 -.206
COMPOUND			
0 (0)	Aft	±.15	.253 -.262
	Fwd	±.15	.358 -.157
20.6 (40)	Aft	±.15	.152 -.364
	Fwd	±.15	.273 -.241

FIGURE 3-38 LONGITUDINAL ACCELERATION CAPABILITY

The helicopter and compound transmissions are designed to sustain a system lift to gross weight (F/W) ratio of 1.05 with the loss of one engine. Thus, both aircraft meet the level 1 requirement and exceed the level 2 requirement.

3.4.3.2 VTOL Approach

Controlled VTOL approach capability was investigated for a range of approach speeds of 40 to 100 knots with a 2000 fpm rate of descent while simultaneously decelerating along the flight path in a 25-knot cross wind. The helicopter and compound can meet the requirement of decelerating along the flight path at .15g up to a speed of 48 to 50 knots, respectively. Above these speeds, the steady state deceleration capability is decreased to about .075g at 100 knots. This deficiency could be corrected with the addition of aerodynamic speed brakes.

With both configurations,there is ample collective pitch remaining to produce a normal acceleration of ±.1g with collective pitch only in less than .5 second. The response of rotary wing aircraft to a collective pitch input is nearly instantaneous.

3.4.4 VTOL Control Systems Lags

The angular acceleration response of the helicopter to -3 degree longitudinal step input is shown in Figure 3-39. As can be seen, the peak acceleration (pitch moment) is reached in .2 second, thus surpassing the requirement of Reference 2. This is typical of the angular responses of both aircraft about each axis for all speeds.

The normal load factor response of the helicopter is shown in Figure 3-40 for a 2-degree step collective input. The maximum normal load factor is obtained in .4 second, again surpassing the requirements of Reference 2. This is a typical response for both aircraft.

3.4.5 Hovering and Low Speed Stability

The hover stability of the helicopter and compound is shown in Figure 3-41 for the most critical center-of-gravity position. As can be seen, both aircraft meet the level 1 requirement. The attitude, pitch rate, and velocity feedback gains required are typical of those used on present helicopters.

The triply redundant automatic stability augmentation system provides for continued concurrence with the level 1 requirement even after failure of any one control system element.

3.4.6 Tail Rotor Loss

The vertical tail surfaces on the helicopter and compound were designed for continued flight following loss of anti-torque thrust, for the most critical (aft) center-of-gravity position. At this condition, both aircraft have sufficient directional stability to maintain level flight throughout a 20-knot speed range at roll angles below 10 degrees and sideslip angles below 20 degrees.
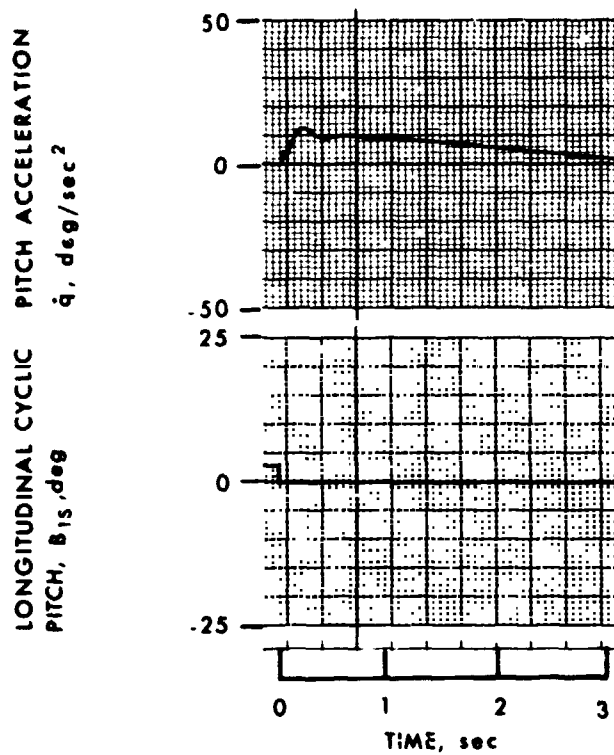
FIGURE 3-39. HELICOPTER – ANGULAR ACCELERATION RESPONSE TO A -3-DEGREE
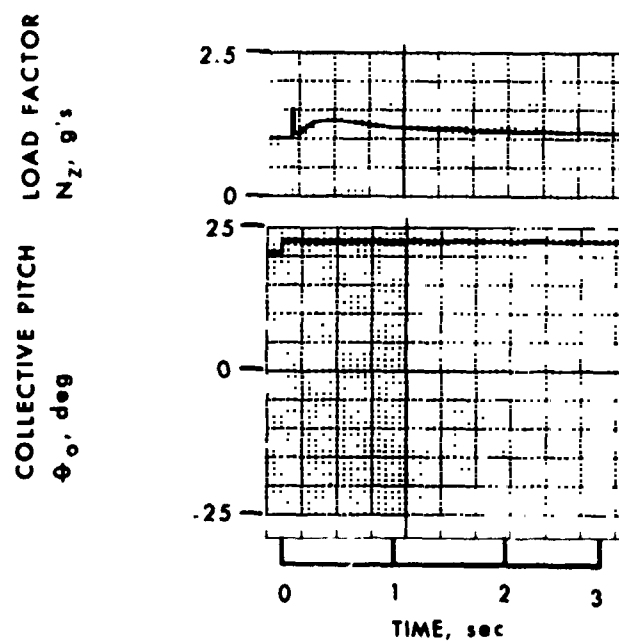LONGITUDINAL STEP INPUT

FIGURE 3-40. HELICOPTER – NORMAL LOAD FACTOR RESPONSE TO A -2 DEGREE
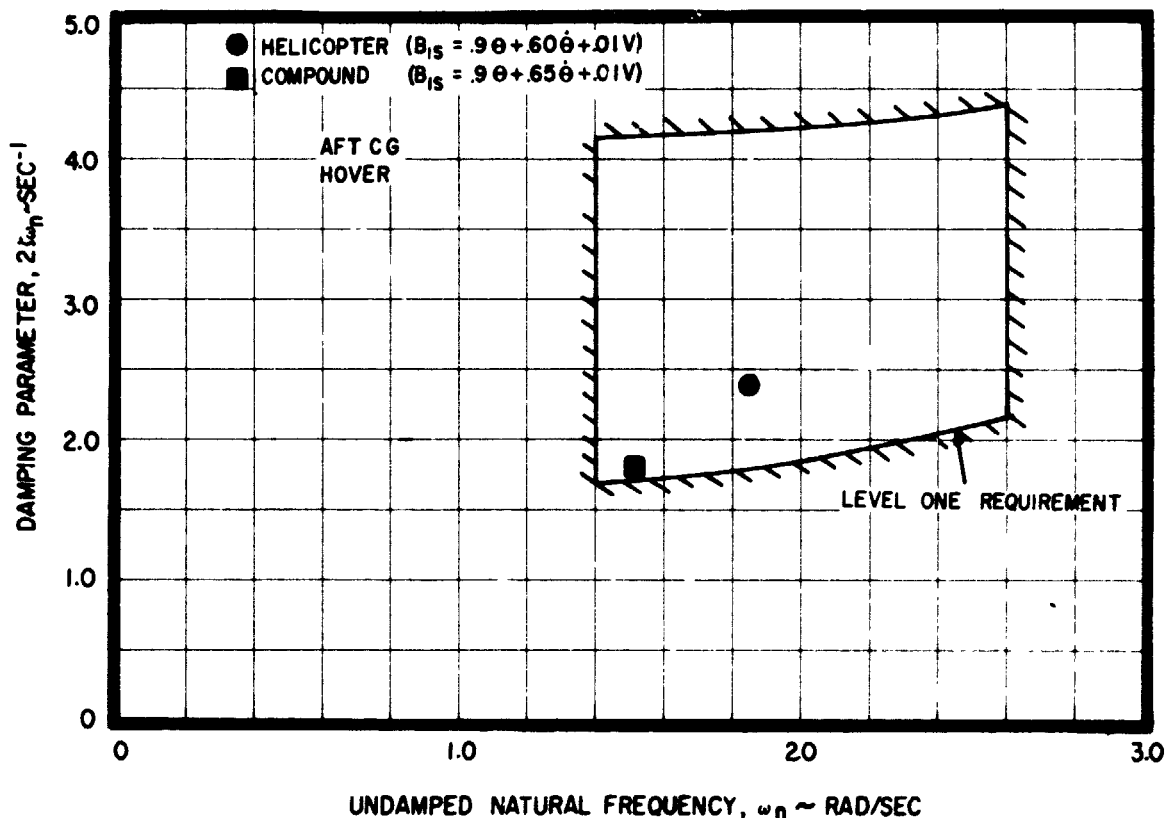COLLECTIVE STEP INPUT

FIGURE 3-41. SHORT PERIOD LONGITUDINAL CHARACTERISTIC ROOTS

If the tail rotor and tail rotor gearbox should separate from the aircraft, the center of gravity will shift forward about 25% of the total range for both aircraft. This shift in center-of-gravity would require only minimal change in longitudinal cyclic trim because the loss of mass is offset by loss in tail rotor lift.

It has been demonstrated in the fixed-base flight simulator that the YUH-60A UTTAS aircraft response to a tail rotor loss is controllable by a pilot, and a safe landing can be made. Both baseline aircraft will respond in similar fashion.

3.4.7 VTOL Take-Off and Landing

Responses to a 5-second, 15 ft/sec longitudinal and lateral gust were investigated for both baseline aircraft in a hover with the automatic flight control system on. Attitude displacements were stable and tended to return to their original trim values without pilot inputs.

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The Sikorsky CH-53E automatic flight control system was used in this study, ;h only the longitudinal gains modified to those given in section 3.4.5. The itrol system used to stabilize the aircraft will not be affected by the loss an engine or a single-component failure in the automatic flight control item.

.8 Cruise Stability

The helicopter and compound were designed to have positive maneuve. stabi- ;y at aft center-of-gravity without stability augmentation. The longitudinal :lic pitch per g ratio is shown in Figures 3-42 and 3-43 for the helicopter l compound. For both aircraft, the neutral point lies behind the aft center- gravity limit, thus exceeding the requirements of Reference 2. The large ·izontal tails result from the center-of-gravity range of both configurations .ng aft of the main rotor shaft. The destabilizing effect of the nac·lles ·ther increases the compound horizontal size.

.9 Attitude Change in Normal Operation

Trim data indicated that the fuselage deck angle of both configurations .1 not exceed 20 degree nose up or be less than -10 degrees nose down.

.10 Force Change in Normal Operation

The force changes on the passengers due to a normal maneuver depend on .ot technique. As long as the aircraft are handled with smooth pilot control)uts, the force change should not exceed those specified, although the pilot l exceed those forces with the remaining control available at some trim iditions.

.11 Ride Qualities in Turbulence

The gust sensitivities of the two aircraft are shown in Figure 3-44 for :ir cruise altitudes and an altitude of 10,000 feet. The compound requires ! addition of a gust alleviation system to soften the ride. The ailerons l be coupled collectively to normal load factor. The weight penalty for :h a system is estimated to be 40 pounds.
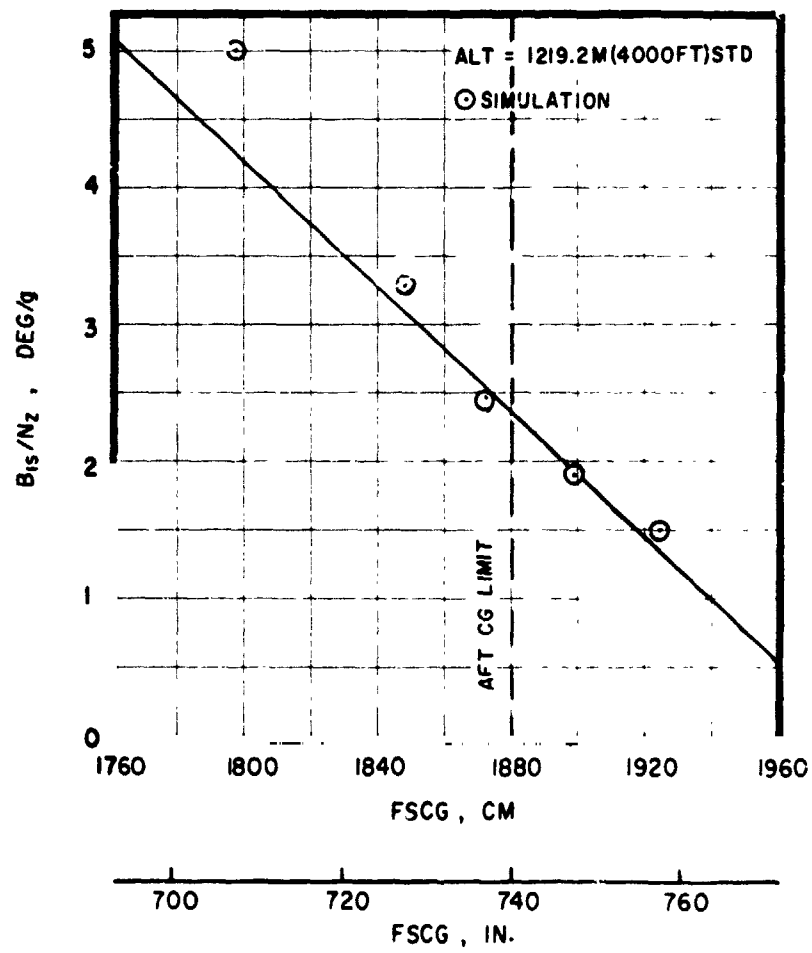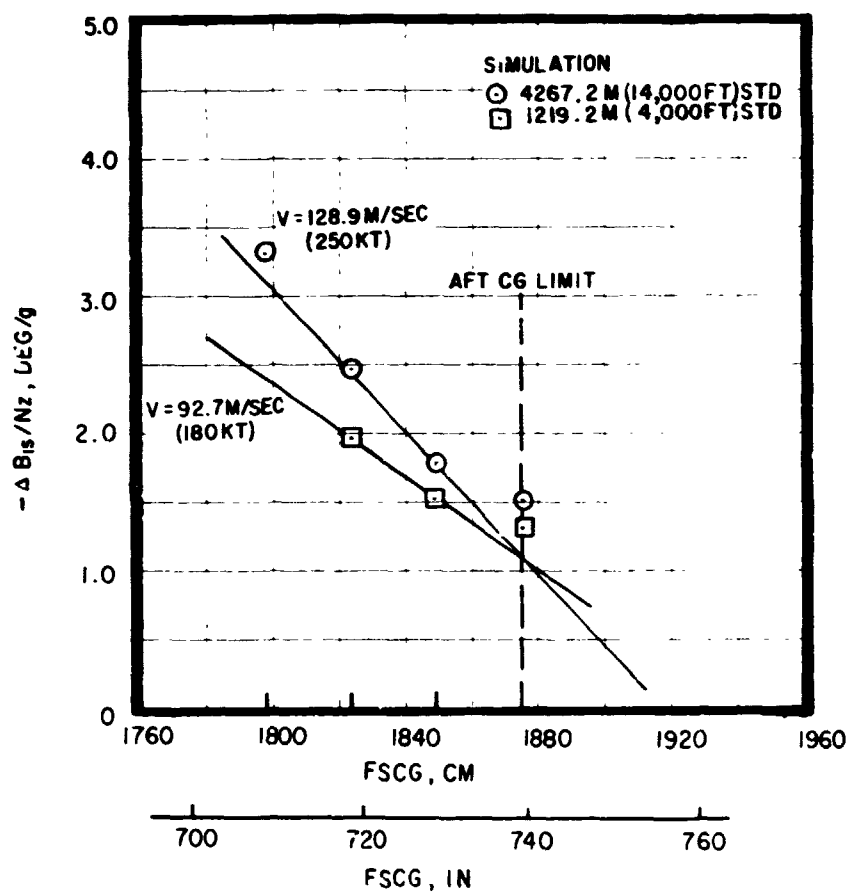
FIGURE 3-42. HELICOPTER LONGITUDINAL STATIC STABILITY

FIGURE 3-43. COMPOUND LONGITUDINAL STATIC STABILITY

	GUST ALLEVIATION FACTOR		A_n/U_{d_e} (g/fps)		
	ROTOR	BODY	ALTITUDE	DESIGN LIMIT	ACTUAL
HELICOPTER	.6	–	4000 ft	.012	.0119
			10000 ft	.018	.0099
COMPOUND	.6	.83	14000 ft	.0228	.0256
			10000 ft	.018	.0291

FIGURE 3-44.   BASELINE AIRCRAFT GUST SENSITIVITY

3.5 Baseline Aircraft Noise Characteristics

3.5.1 Internal Noise - Helicopter

Figure 3-45 presents the cabin soundproofing requirements for the baseline helicopter in hover and cruise. It is clear that gear damping and transmission isolation are necessary to reduce required soundproofing weight to acceptable levels.

It is also apparent from Figure 3-45 that the requirement for 70 dB PSIL in cruise determines the soundproofing weight. To meet this requirement, soundproofing of 0.28 kg/m^2 (0.56 psf) is required, while only 0.25 kg/m^2 (0.5 psf) is required to meet 75 dB PSIL in take-off. This assumes use of the advanced technology soundproofing material discusssed in Section 2.2.1.

Figure 3-46 indicates how soundproofing requirements are incrementally reduced. Notice that, as Figure 3-45 shows, even though damped gears and an isolated transmission are used, soundproofing provides the largest portion of the noise reduction.

3.5.2 Internal Noise - Compound

The internal noise of the compound helicopter in cruise is dominated by the noise of the auxiliary propulsors because little power is being transmitted through the main transmission during cruise. Bare cabin levels of more than 95 dB PSIL are generated by the auxiliary propulsors while levels below 85 dB PSIL result from the transmission (assuming damped gears and isolation). The required soundproofing to meet the specified 70 dB PSIL in cruise was shown in Figure 3-22 for propellers and fan engines. The baseline design uses fan engines as auxiliary propulsors separated by one fan diameter from the fuselage, requiring an average soundproofing density of 0.34 kg/m^2 (0.7 psf). The soundproofing to meet 75 dB PSIL in take-off is only 0.2 kg/m^2 (0.6 psf).

3.5.3 External Noise

The 150-meter (500-foot) sideline noise for the hovering vehicles was calculated using the techniques described in Reference 16, as shown in Figures 3-47 and 3-48 for the helicopter and the compound, respectively. The helicopter generates 93.5 PNdB at 150 meters (500 feet), while the compound produces 95.2 PNdB. The main and tail rotors dominate the spectrum on both vehicles.

Typical take-off and landing profiles were calculated with aid of the Low Speed Dynamic Performance program, Reference 17. The results were used as input to the V/STOL Noise Model (Reference 12) to calculate ground noise contours. Figures 3-49 and 3-50 show the 90, 95, and 100 PNdB contours for the helicopter and compound take-offs, and Figures 3-51 and 3-52 show the landing contours. As expected, the compound contours are larger because of the flatter take-off profile for this type of aircraft.

The study conducted by Munch and King and cited earlier, developed noise criteria for the acceptance of helicopters by communities. This study concluded
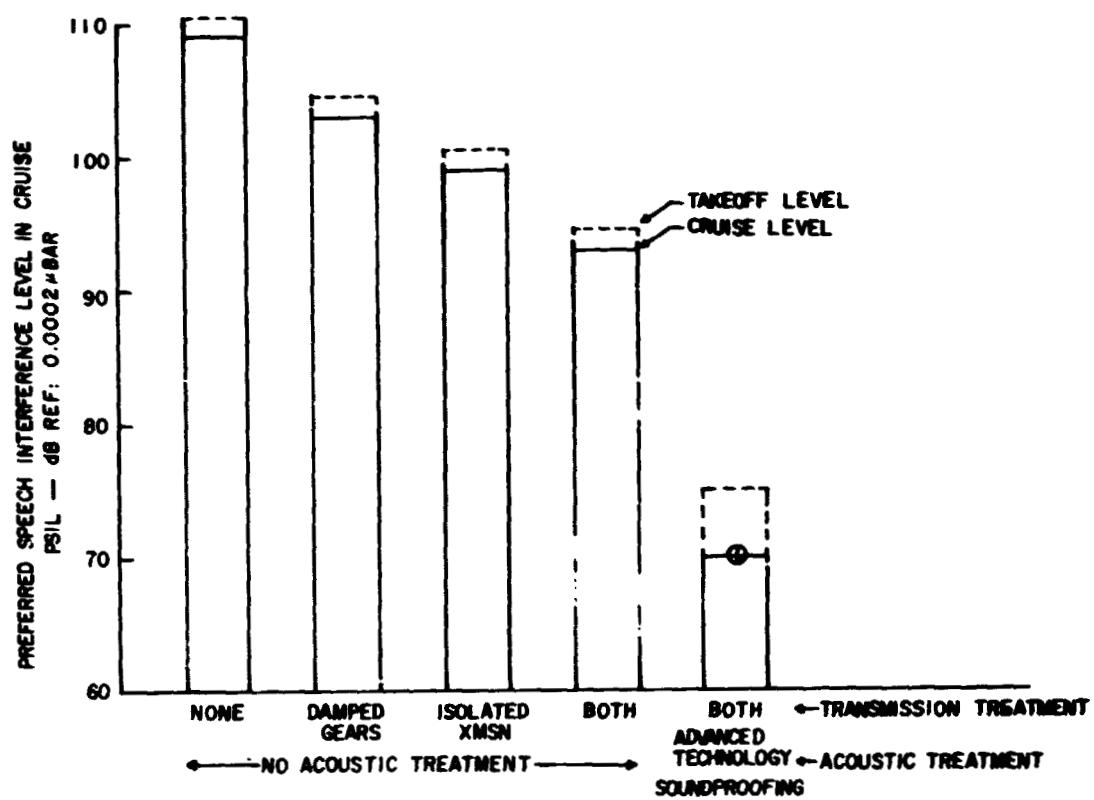
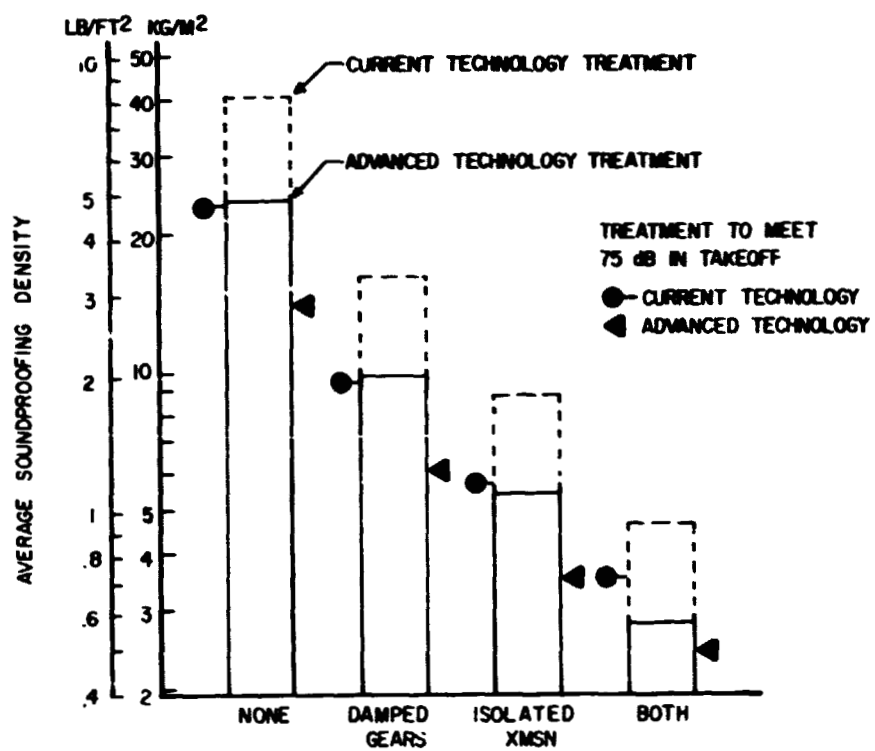FIGURE 3-43. HELICOPTER - INTERNAL ACOUSTIC TREATMENT

60

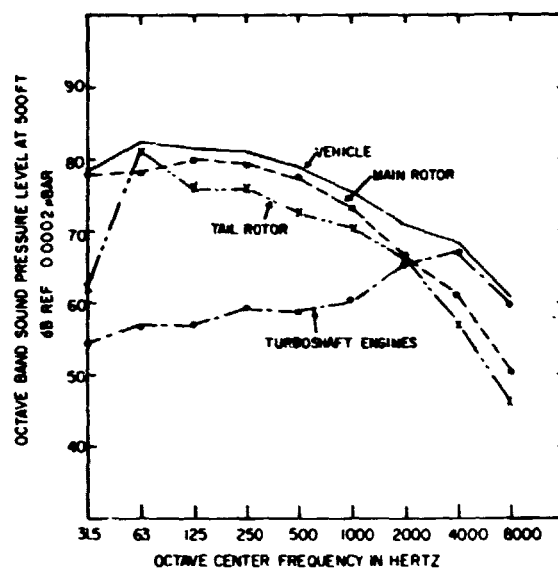FIGURE 3-46.   HELICOPTER - CABIN SOUNDPROOFING DENSITY

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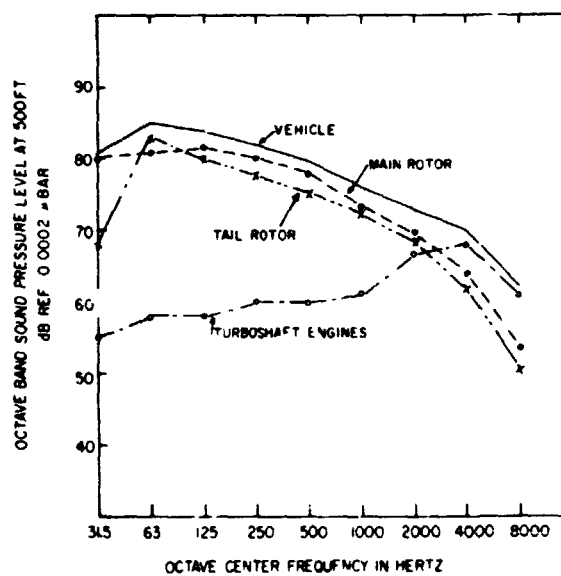FIGURE 3-47. BASELINE HELICOPTER EXTERNAL NOISE SPECTRUM

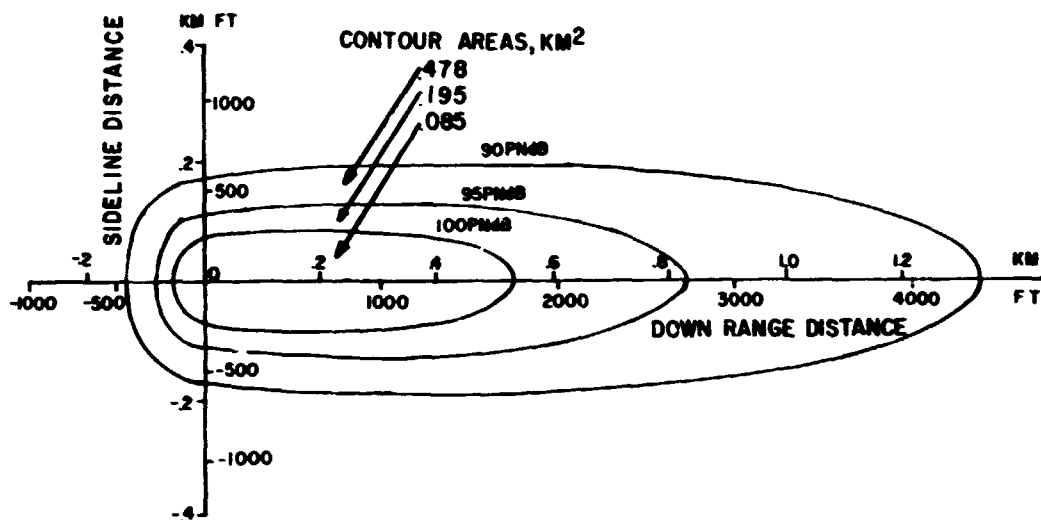FIGURE 3-48. BASELINE COMPOUND EXTERNAL NOISE SPECTRUM

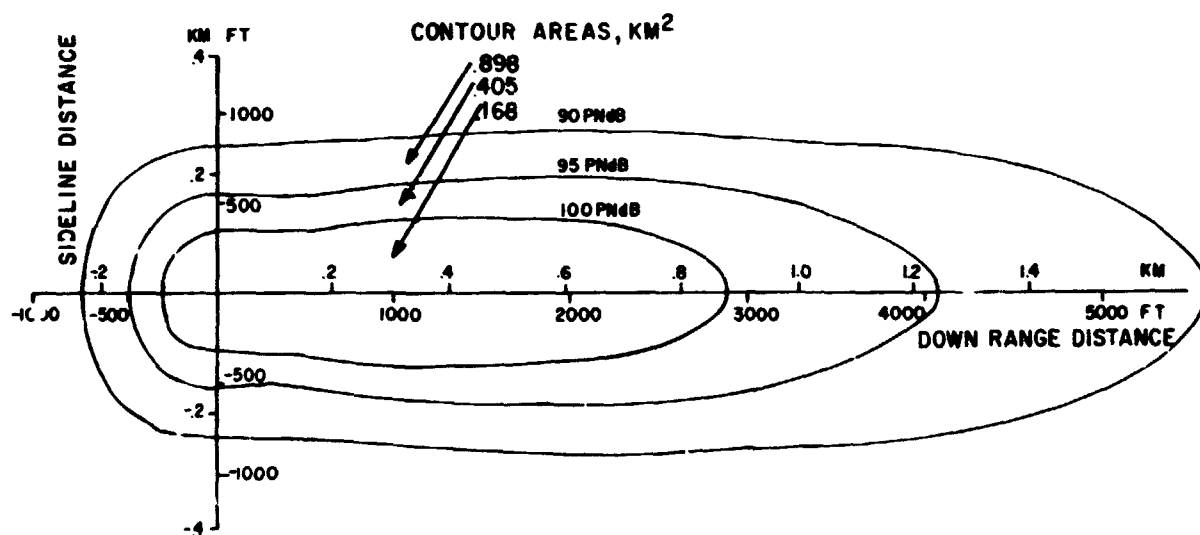FIGURE 3-49. BASELINE HELICOPTER TAKE-OFF PNL CONTOURS

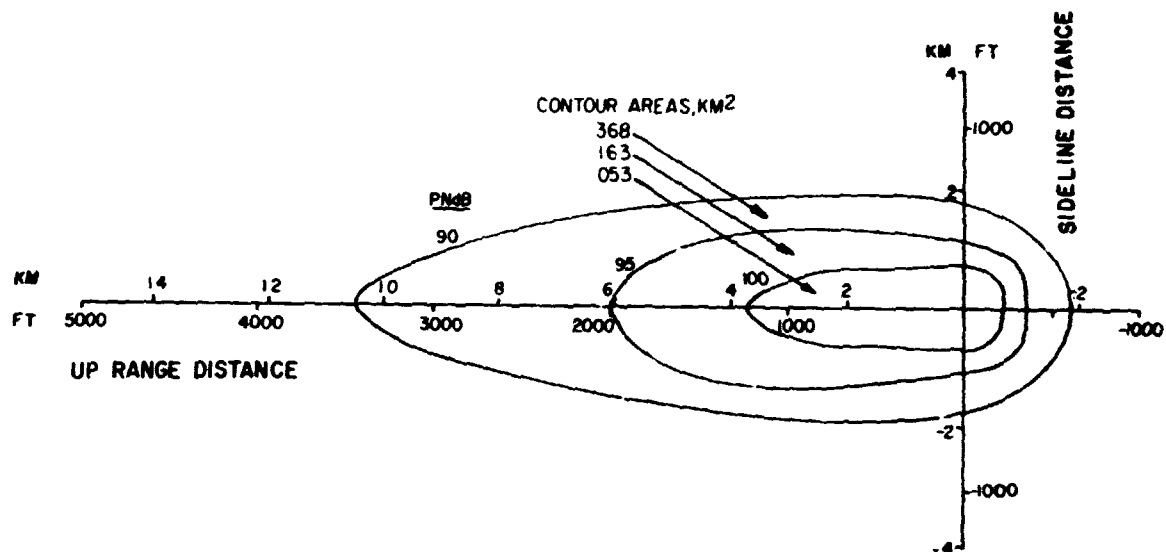FIGURE 3-50. BASELINE COMPOUND TAKE-OFF PNL CONTOURS

63

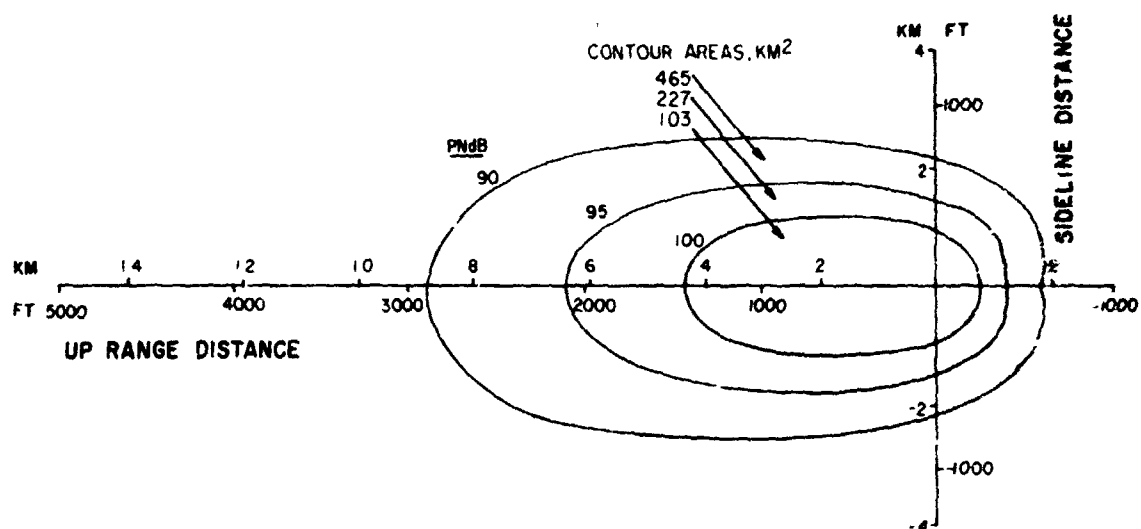FIGURE 3-51.   BASELINE HELICOPTER LANDING PNL CONTOURS

FIGURE 3-52.   BASELINE COMPOUND LANDING   PNL CONTOURS

that an L_{DN} (Day-Night noise level) criterion based on local ambient noise was more meaningful than a single number criterion such as PNL. The L_{DN} measure takes account of the number of operations per day or night, the flight path, the aircraft sound level, and the ambient noise level. The aircraft sound level is specified in terms of the Single Event Noise Exposure Level (SENEL) in dBA. This unit is equivalent to EPNL in that it is the duration and frequency corrected dBA level.

The helicopter and compound SENEL take-off noise contours are shown in Figure 3-53 and 3-54 and compared with the Reference 1 community acceptance criteria in Figure 3-55. The locations shown in Figure 3-55 refer to the typical heliport locations and operations discussed in Reference 1. The details are shown in Figure 3-56.

3.6 Performance

The baseline helicopter and compound are designed to operate in the 1985 commercial environment. Both aircraft can hover out of ground effect at sea level 90 degrees Farenheit conditions, at no more than 109% of take-off power with one engine incperative. Cruise speed for the helicopter is 89 m/sec (173 knots); for the compound it is 129 m/sec (250 knots). These speeds are achieved at no more than maximum continuous engine power.

3.6.1 Aircraft Power Requirements and One-Engine-Inoperative Capabilities

Helicopter power required versus airspeed is shown in Figure 3-57. Low speed performance at sea level 90 degrees and cruise performance at 1219 m (4000 ft) standard are given. The critical installed power condition is the sea level 90-degree hover out of ground effect with one engine inoperative. This capability provides for safe recovery at any instant during a typical take-off procedure following malfunction of one engine.

Compound baseline power required versus airspeed is presented in Figure 3-58. The critical engine sizing condition is at the cruise speed with maximum continuous power. Again, because more than enough power is installed to allow hover OGE at the sea level 90 degree point after loss of one engine, a safe recovery can be made during a typical take-off procedure following a single engine malfunction.

The mission profiles were shown in Figure 2-3. Figures 3-59 and 3-60 show the mission analysis output from the design model computer program.

3.6.2 Autorotation

The rotary wing VTOL offers a safety advantage over other types, because of its autorotative capability to a safe landing with short rolling distance following total loss of power. The autorotative envelopes are shown in Figures 3-61 and 3-62. The helicopter envelope is broader, with lower roll-on speed because of its lower disc loading, and comparatively greater stored rotor energy. These envelopes would be very much broader for the less unlikely condition in which two of the three engines have lost power.
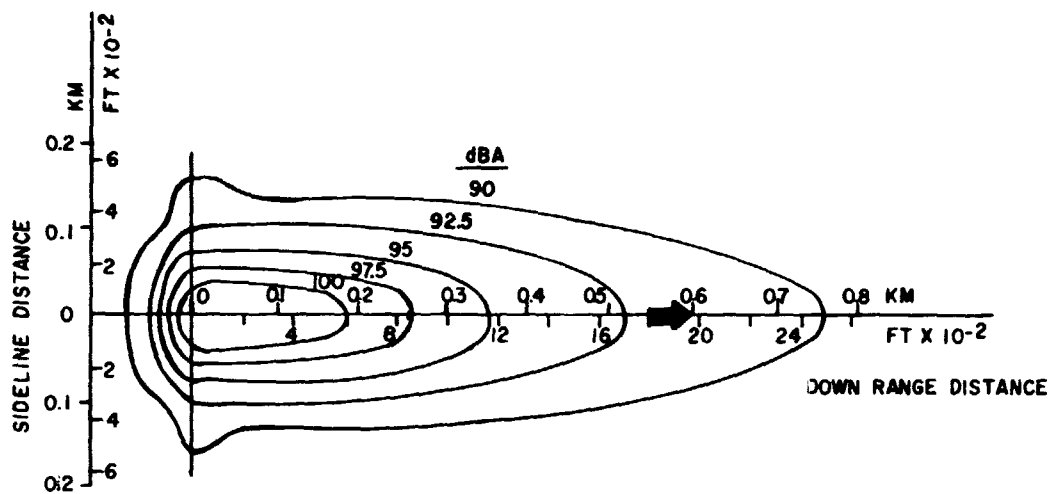
65

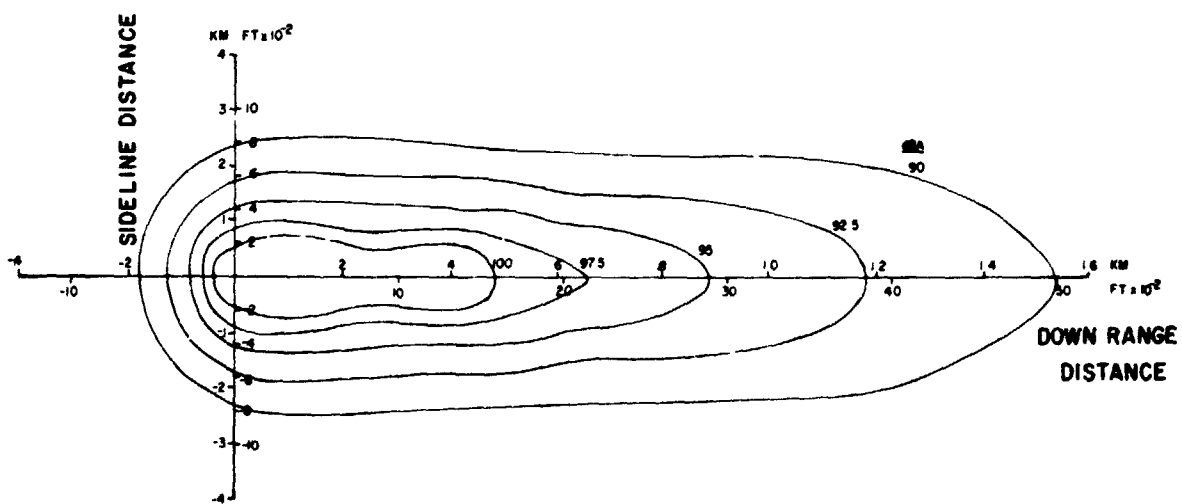FIGURE 3-53. BASELINE HELICOPTER TAKE-OFF SENEL CONTOURS

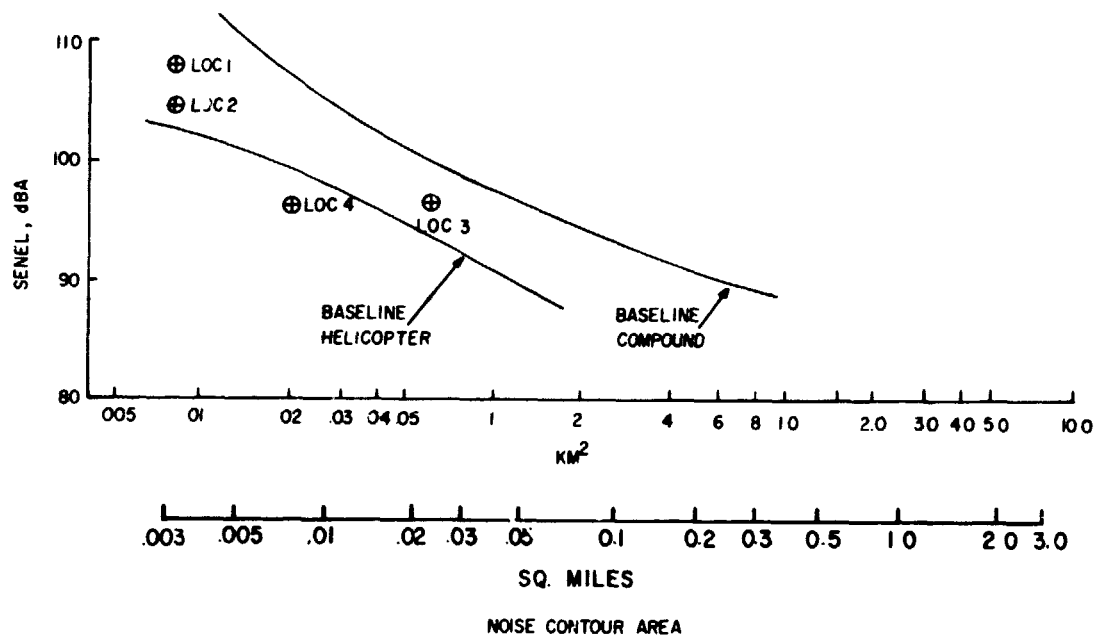FIGURE 3-54. BASELINE COMPOUND TAKE-OFF SENEL CONTOURS

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FIGURE 3-55. TAKE-OFF SENEL COMPARED WITH COMMUNITY ACCEPTANCE CRITERIA

Location	Local Ambient[1] dBA	Number of Operations[2] Daytime	Number of Operations[2] Nighttime	Equivalent[3] Daytime Operations	Allowable L_{DN}/ Footprint Area dBA/Acres
1. City center near highways, river, docks, etc.	80	64	2	84	82/2
2. City center near business district	75	64	2	84	77/2
3. Urban shopping center	65	36	0	36	67/15
4. Urban residential	50	18	0	18	60/5

Notes:

1. Local ambient is defined here as the L_{50} level; that is, the level exceeded 50% of the time.

2. Daytime = 0700 hrs to 2300 hrs
Nighttime = 2300 hrs to 0700 hrs

An operation consists of a takeoff and a landing. Thus the total L_{DN} at a point includes the noise from takeoff and from landings. (For example, if landings are from the West and takeoffs are toward the West, a point under the flight path is exposed to twice as many operations as if the takeoff were toward the East.)

3. A single nighttime event is equivalent to 10 daytime events.

FIGURE 3-56. RECOMMENDED CIVIL HELICOPTER OPERATIONS FOR DETERMINING
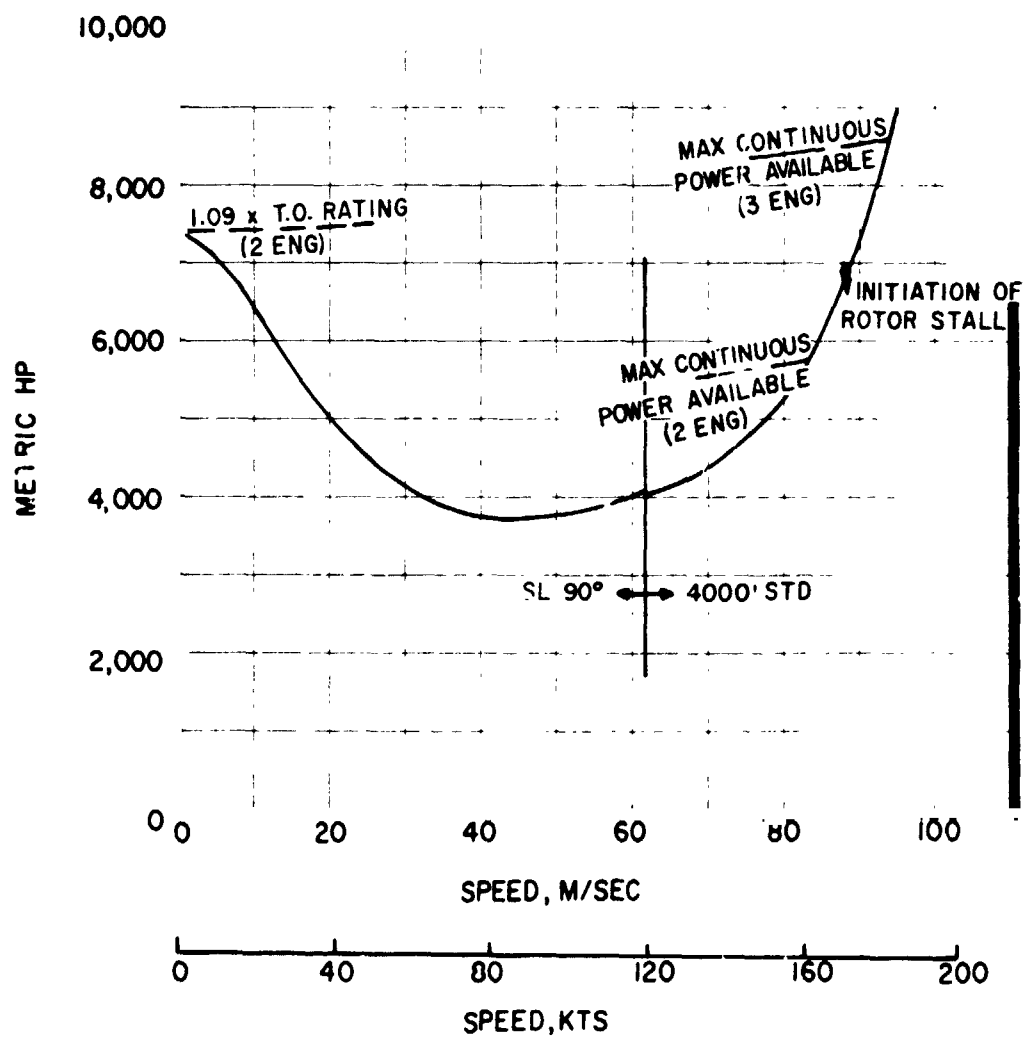AIRCRAFT COMPLIANCE WITH NOISE CRITERIA
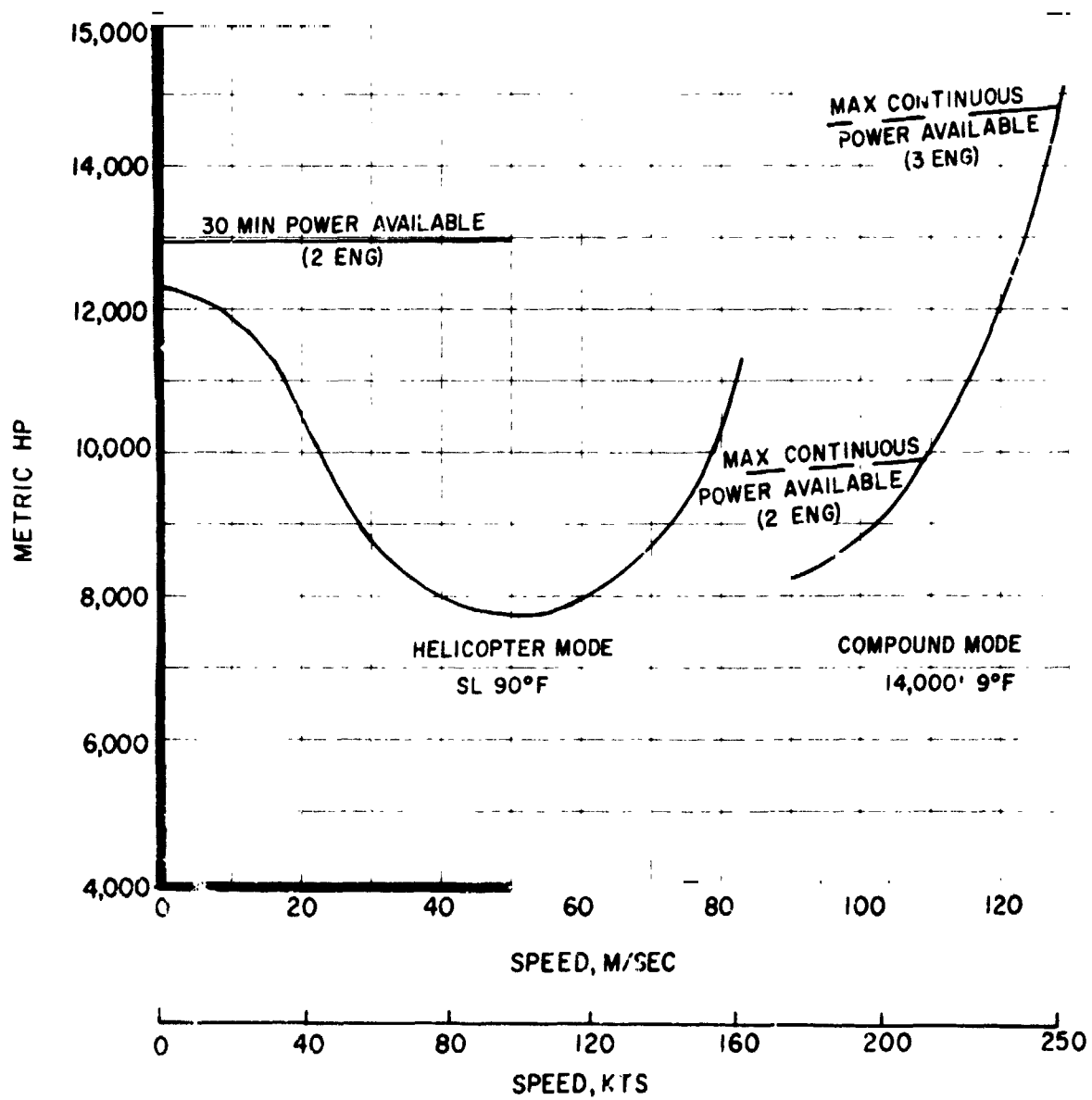
FIGURE 3-57.   HELICOPTER PERFORMANCE SUMMARY

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FIGURE 3-58 COMPOUND PERFORMANCE SUMMARY

```
TOGW= 58136.5 LBS.,   ROTOR RADIUS= 46.10 FT.,    PARASITE DRAG = 39.6 SQ.FT.
```

MODE	GR.WT (LBS)	TEMP (DEG.F)	ALT (FT)	SPEED (KTS)	VSTALL (KTS)	DIST (N.MI)	TIME (MIN)	FL.AR. (SQ.FT)	SHP	FUEL (LBS)	SFC
TAXI	58106.	90.	0.	--	--	--	1.0	--	8611.7	66.5	.4214
HOVER	58063.	90.	0.	--	--	--	.5	--	7304.5	26.0	.4268
MANUVR	58042.	0.	0.	180.0	188.1	.8	.5	39.60	4164.0	16.7	.4796
ACCEL	58025.	90.	0.	120.0	188.1	1.0	.5	39.60	4384.5	17.2	.4710
CLIMB	57850.	52.	2000.	140.0	178.7	18.7	8.0	39.60	5821.2	331.8	.4275
CRUISE	57023.	45.	4000.	172.8	172.8	85.2	29 -	39.60	6474.1	1320.2	.4137
CRUISE	55705.	45.	4000.	174.6	174.6	85.2	29.3	39.60	6517.9	1319.7	.4133
DESCNT	54992.	48.	3000.	150.0	178.9	14.0	4.0	39.60	3663.2	115.3	.720
MANUVR	54911.	52.	2000.	100.0	181.8	2.5	1.5	39.60	4004.4	47.1	.4709
DESCNT	54861.	53.	1500.	80.0	183.0	2.7	2.0	39.60	3035.6	52.9	.5179
DESCNT	54808.	57.	500.	80.0	185.5	2.7	2.0	39.60	3006.6	52.8	.5267
TAXI	54757.	90.	0.	--	--	--	1.0	--	6889.9	49.9	.4309
RESERVE-LOITER	54927.	45.	4000.	89.6	176.3	29.9	20.0	39.60	3954.0	609.6	.4625
CRUISE	53767.	45.	4000.	153.9	177.1	50.0	19.5	39.60	508?	710.7	.4301

```
TOTAL MISSION FUEL IS   4724.3 LBS

TOTAL MISSION TIME IS    77.9 MINS
```

FIGURE 3-59.   HELICOPTER MISSION ANALYSIS

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MODE	GR.WT (LBS)	TEMP (DEG.F)	ALT (FT)	SPEED (KTS)	DIST (N.MI)	TIME (MIN)	FL.AR. (SQ.FT)	SHP	FUEL (LBS)	SFC
TAXI	75263.	0.	0.	--	--	1.0	--	17498.5	125.9	.4219
HOVER	75778.	90.	0.	--	--	.5	--	12102.1	44.7	.4435
MANUVR	75 .	52.	2000.	100.0	.8	.5	41.29	5089.8	23.5	.5548
ACCEL	75711.	52.	2000.	140.0	1.2	.5	41.29	6289.7	26.6	.5089
CLIMB	75500.	30.	8000.	180.0	13.1	4.4	41.29	13650.6	407.1	.4095
CRUISE	74239.	9.	19000.	250.0	86.9	20.7	41.29	19681.0	2119.8	.4171
CRUISE	72129.	9.	19000.	250.0	86.9	20.7	41.29	19613.8	2103.7	.4168
DESCN	70950.	30.	8000.	200.0	13.3	6.0	41.29	9301.1	261.2	.4213
MANUVR	70785.	52.	2000.	100.0	2.5	1.5	41.29	9749.6	67.7	.570
DESCNT	70739.	53.	1500.	80.0	1.3	1.0	41.28	3192.1	35.5	.6
DESCNT	70683.	57.	500.	80.0	2.2	1.7	41.28	3089.3	66.5	.6197
TAXI	70548.	90.	0.	--	--	1.0	--	19270.8	102.9	.4309
RESERVE- LOITER	70111.	42.	5000.	172.3	39.1	20.0	41.28	9827.8	860.3	.5396
CRUISE	69169.	41.	5000.	190.0	50.0	15.8	41.28	8767.1	1015.2	.4400

TOTAL MISSION FUEL IS 7212.7 LBS

TOTAL MISSION TIME IS 55.5 MINS

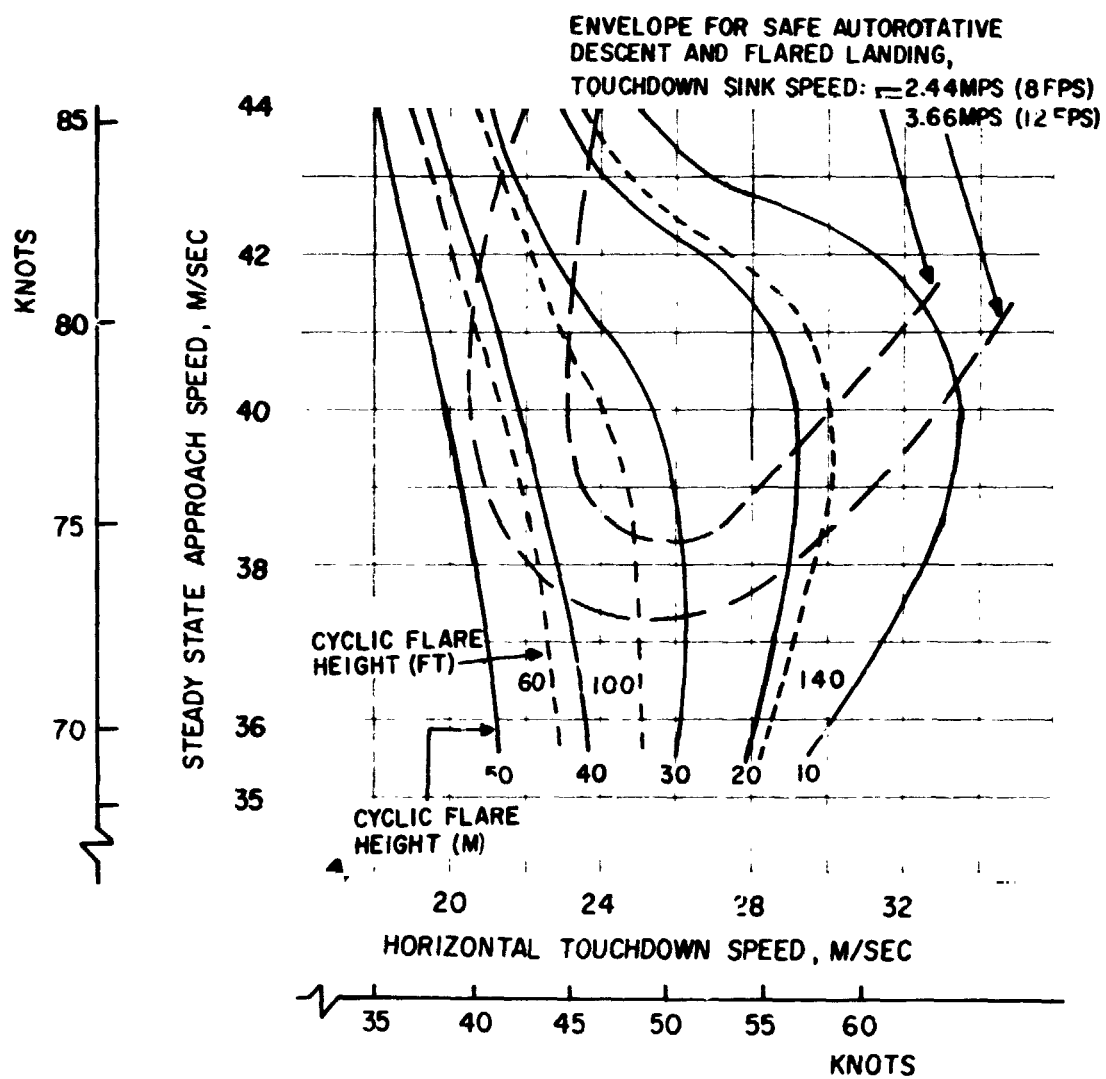FIGURE 3-60. COMPOUND MISSION ANALYSIS

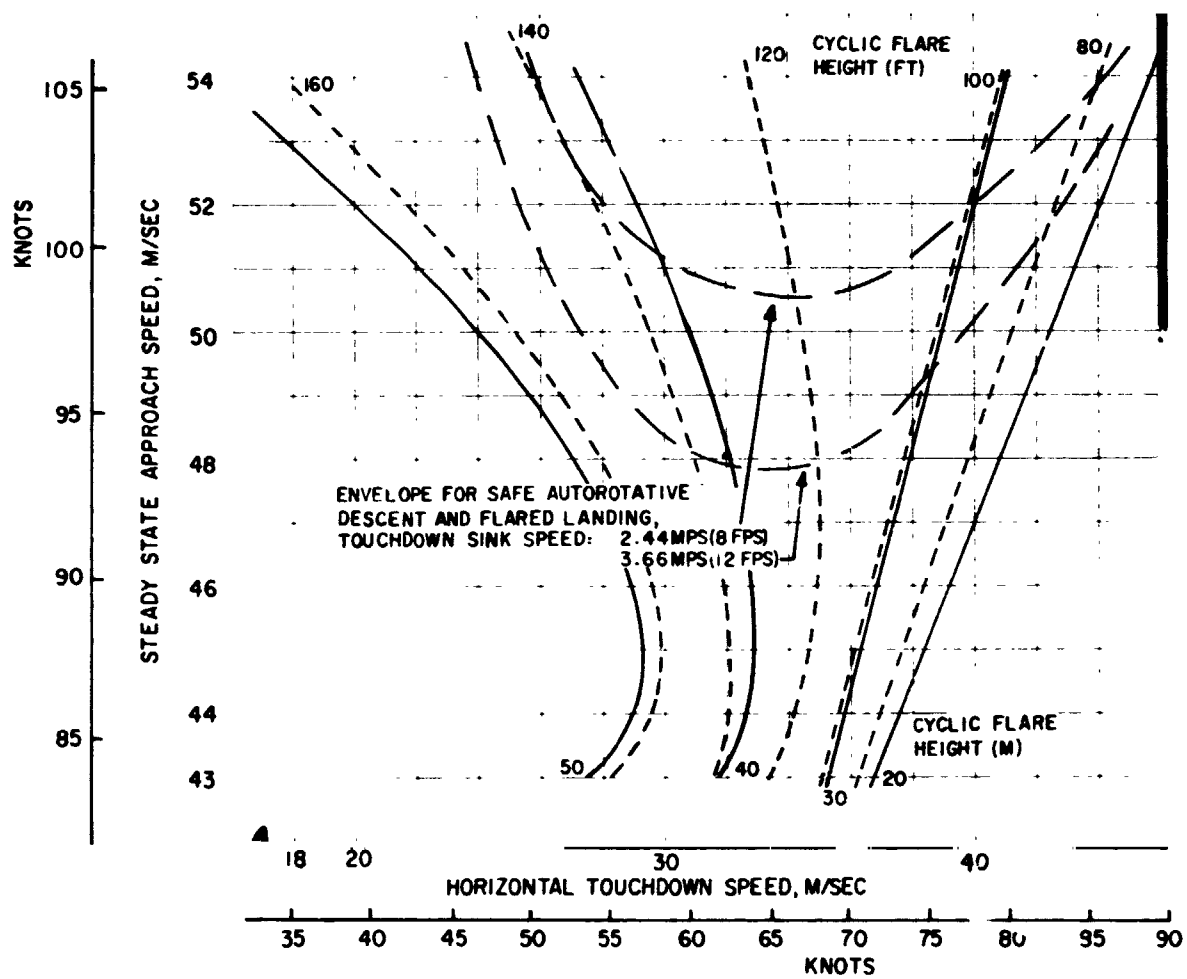71

FIGURE 3-61. HELICOPTER AUTOROTATION ENVELOPE

FIGURE 3-62. COMPOUND AUTOROTATION ENVELOPE

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3.7 Configuration Trade-Offs

3.7.1 Helicopter

In selecting the baseline helicopter, eight configuration trade-offs were conducted. These are summarized in Figure 3-63. In all cases, except the V-tail, the alternative configuration features increased the direct operating cost and therefore were not incorporated in the baseline design.

Elastomeric Main Rotor Head

The elastomeric rotor head is currently being developed to provide a non-lubricated head. At this time, the added weight associated with the elastomeric bearings and retention for large numbers of blades (more than 4) overcomes the reduced maintenance advantage in computing DOC. As the technology is developed, this result may change.

No Rotor Head Fairing

Removal of the rotor head fairing saves weight, but adds parasite drag. Saving in weight empty is offset by the increase in fuel.

Pusher Tail Rotor

When reconfiguring the tail rotor to the left side of the aircraft, the pylon must be canted to allow for clearance. Tail rotor blockage is decreased, but pylon area is increased to obtain the same equivalent vertical empennage effectiveness.

V-Tail with Pusher Tail Rotor

This empennage configuration offers a small DOC advantage over the baseline, because of reduced tail rotor blockage and more aerodynamically efficient tail surfaces. The baseline configuration was preferred,because of the lack of data on a V-tail arrangement.

7- and 8-Abreast Seating

For seven- and eight-abreast seating arrangements, Federal Aviation Regulations stipulate a second aisle, thereby adding further to fuselage section width. Fuselage length is already governed by the rotor size, so the shorter cabin offers no saving in weight. Fuselage drag is increased.

Twin Tail Rotors

Twin tail rotors were considered as a possible candidate feature to reduce external noise. Because the thrust is shared, this arrangement permits a much lower tail rotor disc loading without large increase in size envelope. DOC is increased, because of the added weight of the drive train and tail surfaces.

TRADE	BASELINE	GROSS WEIGHT	DOC RATIO
-		26371 (58137)	1.0
1. ELASTOMERIC MAIN ROTOR HEAD	LUBRICATED HINGE	26708 (58880)	1.014
2. NO ROTOR HEAD FAIRING	FAIRING	26295 (57969)	1.001
3. PUSHER TAIL ROTOR (CH-53E)	TRACTOR (UTTAS)	26401 (58204)	1.001
4. V-TAIL, PUSHER TR	UTTAS STYLE	26316 (58016)	.998
5. 7 ABREAST SEATING, DUAL AISLE	6 ABREAST, SINGLE AISLE	27052 (59639)	1.022
6. 8 BREAST SEATING, ᴧ AISLE	6 ABREAST, SINGLE AISLE	27271 (60121)	1.029
7. TWIN TAIL ROTORS	SINGLE ROTOR	26549 (58530)	1.007
8. FAN-IN-FIN	TAIL ROTOR	26820 (59110)	1.015

FIGURE 3-63. HELICOPTER CONFIGURATION TRADE-OFFS

TRADE	BASELINE	GROSS WEIGHT	DOC RATIO
-		34440 (75926)	1.0
1. ELASTOMERIC MAIN ROTOR HEAD	LUBRICATED HINGE	34806 (76733)	1.011
2. PUSHER TAIL ROTOR (CH-53E)	TRACTOR (UTTAS)	34933 (77014)	1.015
3. V - TAIL, PUSHER TR	UTTAS STYLE	34858 (76849)	1.012
4. 7-ABREAST SEATING, DUAL AISLE	6-ABREAST, SINGLE AISLE	35885 (79112)	1.038
5. 8-ABREAST SEATING, DUAL AISLE	6-ABREAST, SINGLE AISLE	36212 (79833)	1.045
6. SIMPLE WING	FLAPS & LEADING EDGE DEVICES	35235 (77678)	1.026
7. SIMPLE WING + SPOILERS	FLAPS & LEADING EDGE DEVICES	34823 (76770)	1.013
8. TWIN TAIL ROTORS	SINGLE ROTOR	35241 (77692)	1.023
9. FAN-IN-FIN	TAIL ROTOR	34590 (76240)	1.005
10. VARIABLE TWIST * MAIN ROTOR	CONSTANT GEOMETRY	33976 (74903)	0.98

* REPRESENTS DEVIATION FROM STUDY GUIDELINES.

FIGURE 3-64. COMPOUND CONFIGURATION TRADE-OFFS

Fan-in-Fin

The fan-in-fin was evaluated to determine whether an anti-torque device of smaller size, but absorbing more power, would be beneficial. For the helicopter, in which the design hovering point is critical in sizing the installed power, aircraft weight and size are significantly increased. Cruise performance advantages due to drag reduction were small compared with the effects on aircraft size.

3.7.2 Compound

The ten compound configuration trade-offs are summarized in Figure 3-64. In all cases, the features studied resulted in higher direct operating costs, except for the variable twist main rotor, which was not included in the base-line because it represented a deviation from study guidelines. The explanations of the results of trade-offs 1, 2, 3, 4, 5, and 8 are similar to those given for the helicopter. For the fan-in-fin, trade-off 9, the result is more marginal. Hovering inefficiency serves only to increase main gearbox size, while installed power is sized by the cruise requirement.

Simple Wing and Simple Wing plus Spoilers

Flaps and leading-edge devices, employed on the baseline, add complexity and weight. However, a simple wing without flaps adds significantly to vertical drag. Also, it is necessary to spoil wing lift in order to load the rotor, and so maintain rpm, during autorotative descent. Leading-edge devices can also be deployed to reduce vertical drag and would be designed to minimize tail buffet from the shed vortices in forward flight.

Variable Blade Twist

High blade twist desirable for good hover performance and reduction of hover noise signature is not beneficial in high-speed flight, because of high blade stresses and rotor inplane drag forces. The variable twist concept would provide in-flight adjustment to the optimum amount for each phase of the mission. There is a significant benefit to DOC, regardless of the effect on the noise signature. Because the concept represented a deviation from study groundrules, it not included in the baseline design.

3.8 Noise Sensitivities

In order to derive the quiet and noisy members of the helicopter and compound families, it was necessary to assess the change in external noise resulting from design changes to the rotor system. The acoustic sensitivity of the baseline rotor to changes in number of blades, rpm, C_T/σ, disc loading, and twist was determined parametrically. To produce meaningful results for a manageable number of points, the parameters were varied individually around the baseline values.

From Figures 3-65 and 3-66, it is obvious that the main rotor is relatively insensitive to changes in disc loading ar number of blades. This is so, because for this particular rotor, the broadband component of the noise dominates the Perceived Noise Level, and broadband noise is not sensitive to changes in disc loading. As long as C_T/σ is held constant, broadband noise does not change with changing blade number.

The most significant parameter is tip speed. It affects both the rotational and the broadband component of the noise. Twist has limited effect around the baseline design point, because it has little effect on broadband noise. A combination of hanges of parameters such as tip speed, twist, and C_T/σ, must be employed to significantly change main rotor noise.

The sensitivity of helicopter tail rotor noise is shown in Figure 3-67 and 3-68. Here the trend is somewhat different than for the main rotor, because the rotational component of noise dominates over the broadband. Thus, disc loading, twist, and number of blades in addition to tip speed are sensitive parameters, but C_T/σ is not.

Compound main rotor noise sentivities shown in Figures 3-69 and 3-70 are similar to those for the helicopter tail rotor, because the rotational component dominant in this case. For this reason, twist is an especially strong para-, along with tip speed. The sensitivity of disc loading and C_T/σ is not
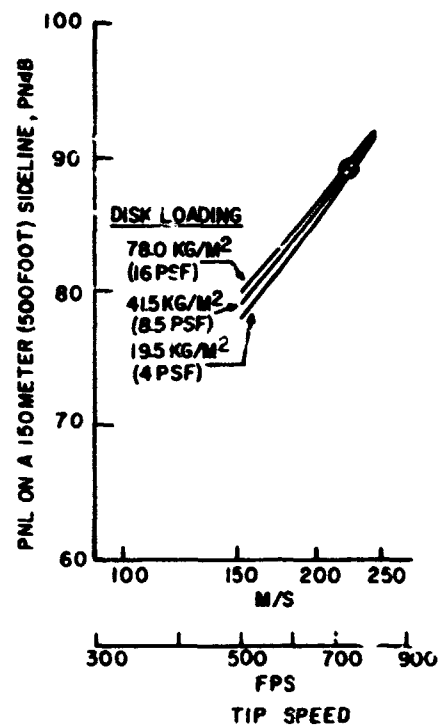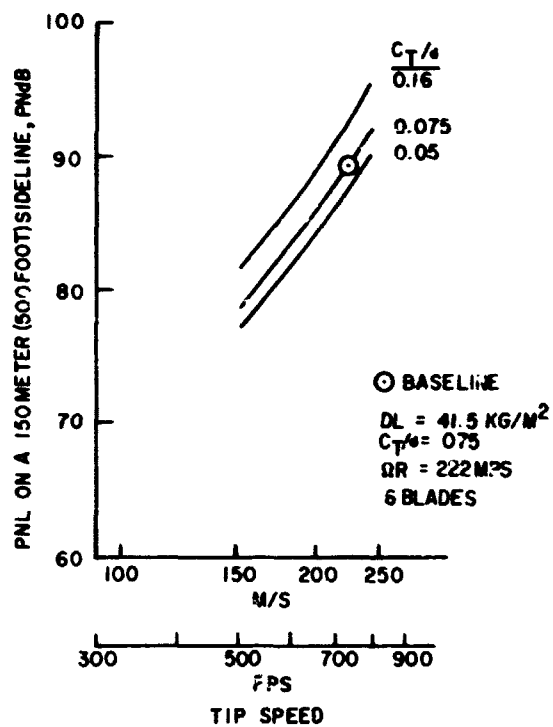
FIGURE 3-65. HELICOPTER - EXTERNAL NOISE vs. MAIN ROTOR TIPSPEED,
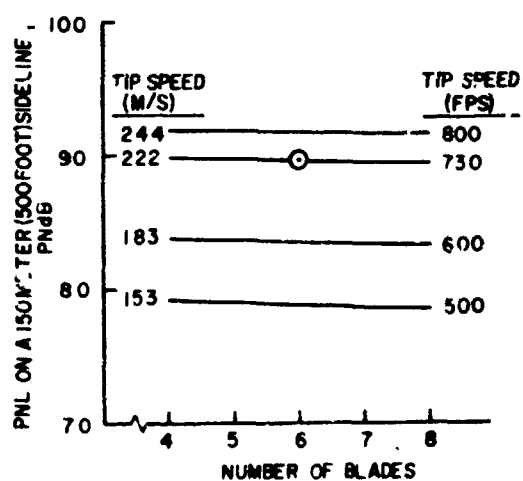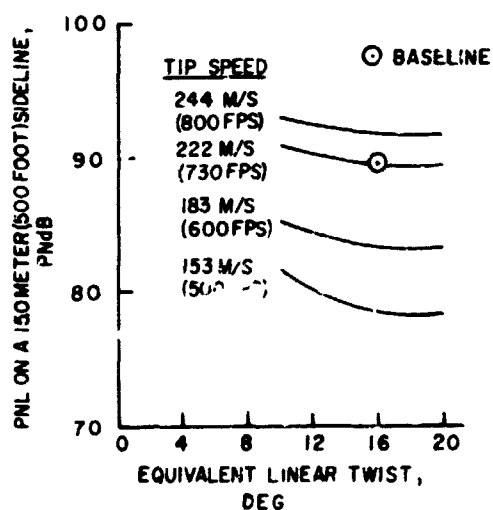C_T/σ, AND DISC LOADING

FIGURE 3-66. HELICOPTER - EXTERNAL NOISE vs. MAIN ROTOR TIPSPEED,
BLADE TWIST, AND NUMBER OF BLADES

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The compound tail rotor parametric noise study, Figures 3-71 and 3-72, shows the dominance of rotational noise for this rotor. It is extremely insensitive to C_T/σ variations, but varies widely with changes in disc loading twist, and tip speed.
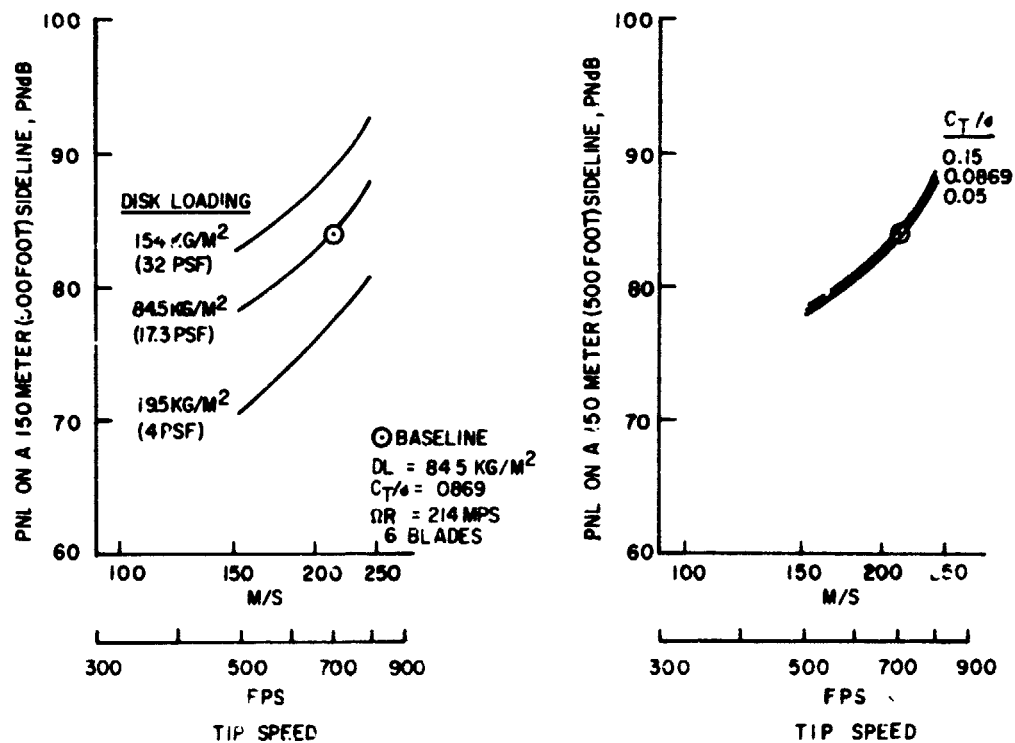

FIGURE 3-67   HELICOPTER - EXTERNAL NOISE vs. TAIL ROTOR TIPSPEED,
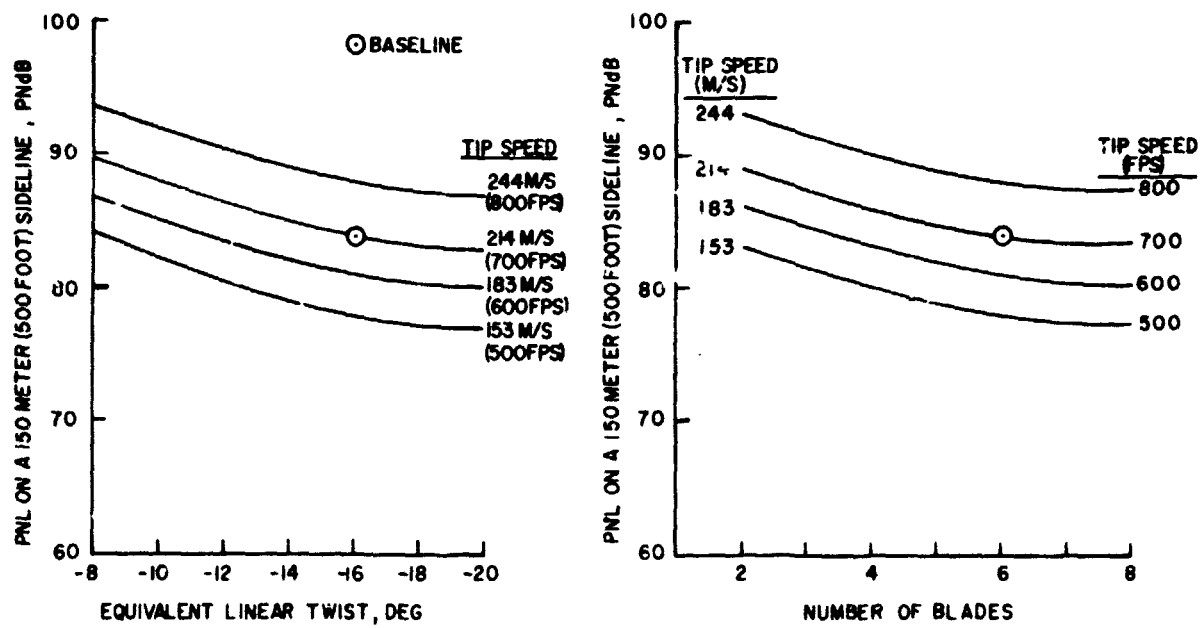   DISC LOADING, AND C_T/σ

FIGURE 3-68. HELICOPTER - EXTERNAL NOISE vs. TAIL ROTOR TIPSPEED, BLADE TWIST, AND NUMBER OF BLADES
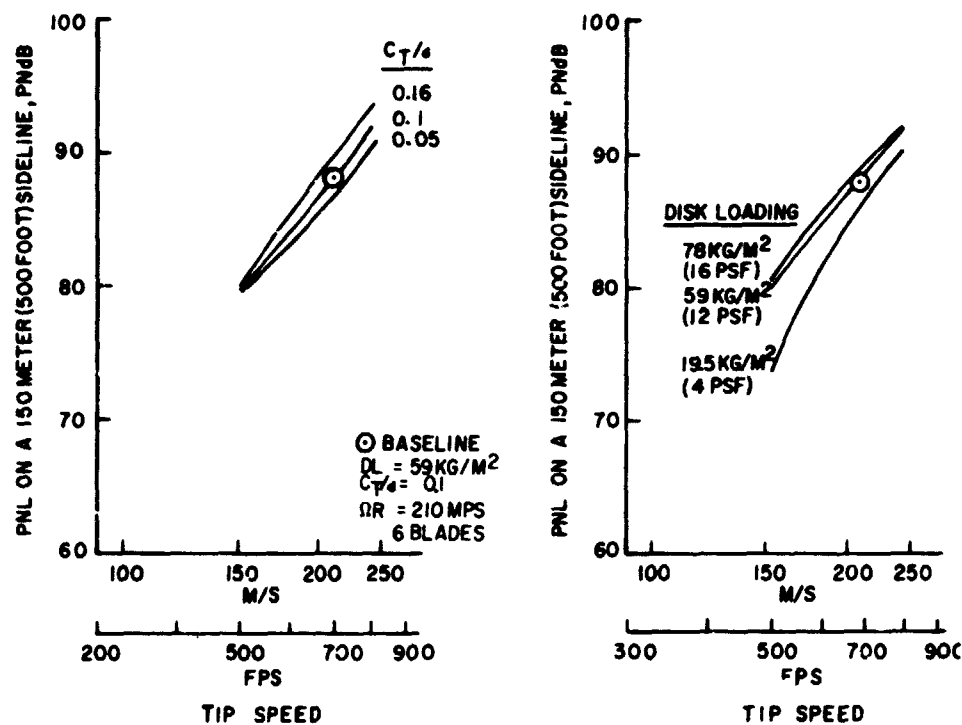
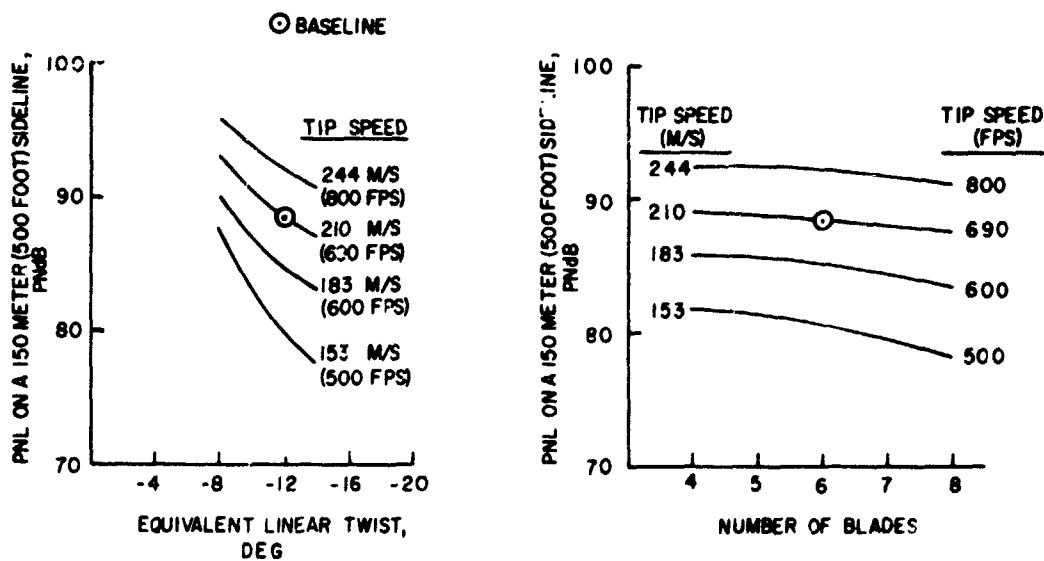FIGURE 3-69. COMPOUND - EXTERNAL NOISE vs. MAIN ROTOR TIPSPEED, C_T/σ AND DISC LOADING

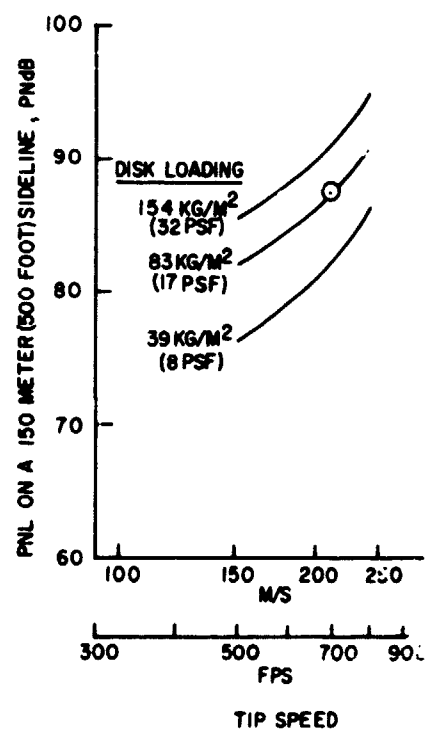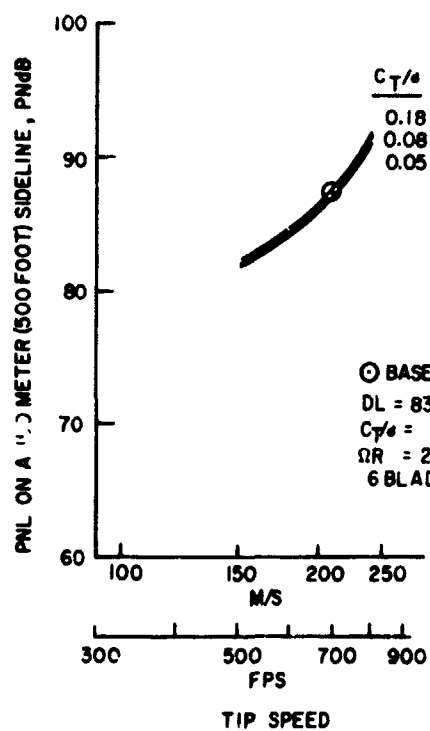FIGURE 3-70. COMPOUND - EXTERNAL NOISE vs. MAIN ROTOR TIPSPEED, BLADE TWIST, AND NUMBER OF BLADES

FIGURE 3-71. COMPOUND - EXTERNAL NOISE vs. TAIL ROTOR TIPSPEED,
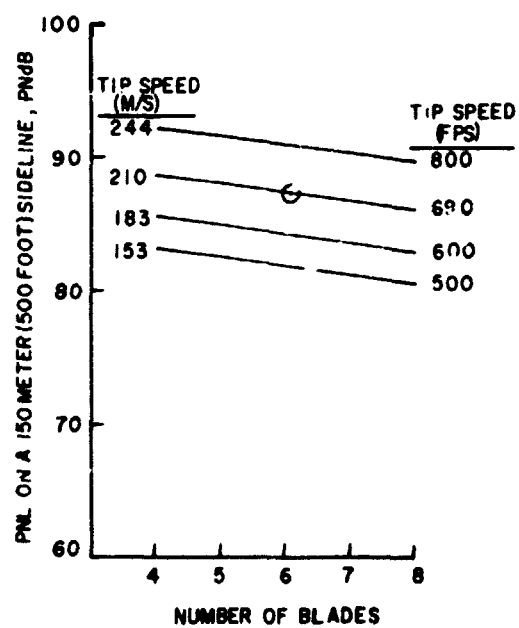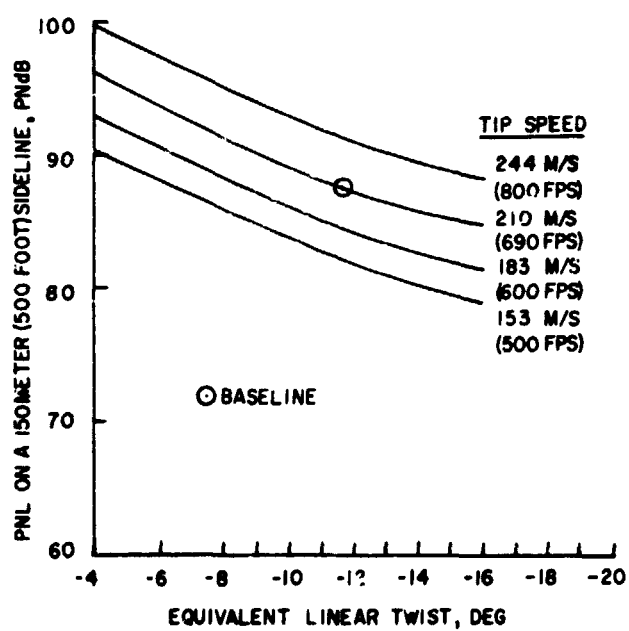C_T/σ, AND DISC LOADING

FIGURE 3-72. COMPOUND - EXTERNAL NOISE vs. TAIL ROTOR TIPSPEED, BLADE TWIST, AND NUMBER OF BLADES

4.0 SENSITIVITY OF DOC TO EXTERNAL NOISE CONSTRAINT

4.1 Technical Approach

Sections 3.3 and 3.8 described the sensitivity of DOC and external noise to variation in the rotor parameters around the baseline values. From these two sets of results, configuration changes summarized in Figures 4-1 and 4-2 for the helicopter and compound, respectively, were evaluated to select changes in design parameters that most significantly affect noise signature but have minimal effect on DOC. Contributions from engines, main rotor, and tail rotor are considered.

4.1.1 Helicopter

Two approaches were employed to achieve a 5 PNdB reduction in external noise. Approach 1 required a 5 dB reduction in all contributors. Main rotor parameters chosen were those that primarily affect the broadband component (tipspeed and C_T/σ). For the tail rotor, emphasis was on rotational noise, (tipspeed and disc loading). The engine noise reduction was accounted for at the rate of 4.54 kilograms (10 pounds) of inlet treatment per dB per engine, Reference 9. Approach 2 requires a minimum reduction in main rotor noise, while tail rotor and engine noise were reduced to a point at which neither contributed significantly to the cumulative noise level.

The 5 dB increase in external noise was achieved by decreasing blade twist from -16 to -10 degrees. This is contrary to the DOC versus twist trend, in that minimum DOC is obtained at -16 degrees, but represents a pre-UTTAS technology aluminum spar blade with decreased manufacturing cost. It was necessary to derive the +5 dB aircraft in this artificial way, because the baseline external noise goal was achieved with the set of rotor parameters that produces minimum DOC. Hover tipspeed was increased to 231.6 m/sec (760 fps). A reduced forward flight tipspeed of 213.3 m/sec (700 fps) was specified in order to achieve the same 173-knot cruise speed as the baseline aircraft. It was assumed that this could be achieved through proper control of the free-turbine rotational speed.

4.1.2 Compound

The 5 dB reduction was again studied using the two-approach system described above, Figure 4-2. Because rotational noise dominates, the effect on noise of varying C_T/σ is less than for the helicopter. The 5 dB increase was obtained by selecting rotor parameters for minimum DOC, that is, main rotor tipspeed increased to 730 fps, twist reduced to -4 degrees, and tail rotor tipspeed increased to 700 fps.

4.2 Results

In obtaining the quiet designs, it was soon apparent that the Approach 2 solutions caused much less degradation in DOC than those of Approach 1, which were discarded. Of the Approach 2 solutions, reduction in main rotor hover C_T/σ from the baseline values (.075 for the helicopter, .1 for the compound) was

		-5 dB A/C		+5 dB A/C
		APPROACH 1	APPROACH 2	
ENGINES	NOISE SIG. CHANGE	-5 dB	-6 dB	0
MAIN ROTOR	NOISE SIG. CHANGE TIPSPEED, FPS C_T/σ	-5 dB 615, 640, 660 .075, .0625, .05	-3 dB 660, 685, 700 .075, .0625, .05	+5 dB 760 .07
	TWIST, DEG	-16	-16	-10
TAIL ROTOR(S)	NOISE SIG. CHANGE TIPSPEED DISC LOADING, PSF	-5 dB 530, 600, 680 17, 12, 8	-9 dB 370, 480, 550 17, 12, 8	+5 dB 730 17

FIGURE 4-1. HELICOPTER EXTERNAL NOISE/DOC TRENDING

		-5 dB A/C		+5 dB A/C
		APPROACH 1	APPROACH 2	
ENGINES	NOISE SIG. CHANGE	-5 dB	-6 dB	0
MAIN ROTOR	NOISE SIG. CHANGE TIPSPEED, FPS C_T/σ	-5 dB 565, 580, 590 .1, .075, .05	-3 dB 610, 625, 640 .1, .075, .05	+5 730 .115
	TWIST, DEG.	-12	-12	-4
TAIL ROTOR/FAN: (1) ROTOR(S)	NOISE SIG. CHANGE TIPSPEED, FPS DISC LOADING, PSF	-5 dB 530, 630, 690 17, 12, 8	-9 dB 350, 500, 580 17, 12, 8	+5 dB 700 17
(2) FAN-IN-FIN	NOISE SIG. CHANGE TIPSPEED, FPS		-9 dB 700	

FIGURE 4-2. COMPOUND EXTERNAL NOISE/DOC TRENDING

C-2

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found to cause large degradation in DOC. Also, minimum DOC was obtained at low tail rotor disc loading and least reduction in tipspeed. At the low disc loadings being considered, however, the tail rotor became unacceptably large in diameter. Therefore, the quiet helicopter solution has two tail rotors, each of low disc loading, with a V-style empennage. Though this style of empennage was discarded in favor of the inverted T for the baseline, section 3.7.1, it is employed for the quiet helicopter because of the geometric compatibility. For the compound, it was found during configuration trade-off studies that the fan-in-fin offered a solution competitive with the tail rotor. The high power consumption of this device in hover, though increasing drive system weight, does not affect engine size when this is being set by the 129 m/sec (250-knot) cruise speed requirement. Because a properly designed fan-in-fin offers a 9 dB external noise reduction over a tail rotor, this device was selected for the quiet compound design. Twenty degrees of thrust deflection, equivalent to tail rotor cant, is achieved with adjustable doors on the downstream side of the fan. Figures 4-3 through 4-6 show the octave spectra of the QH, NH, QC and NC designs at 150 meters (500 feet) to the side of the aircraft. The quiet helicopter signature is now dominated by main rotor noise, while the tail rotor component dominates the spectrum of the noisy helicopter. The main rotor and anti-torque fan contribute equivalently to the quiet compound signature. The noisy compound spectrum is controlled by the main rotor component.

Take-off and landing noise contours were calculated for the four off-design aircraft. The results are shown in Figures 4-7 to 4-10 in terms of PNL, and in Figures 4-11 to 4-14 in terms of SENEL (take-off only). The somewhat greater enclosed areas for the compound are due to the flatter take-off profile this type of aircraft must employ using auxiliary propulsion, in order to avoid negative wing lift and/or high vertical drag penalties. A summary of the contour areas for each PNL is given in Figure 4-15. Enclosed contour areas for specific values of SENEL, compared to community acceptance guidelines are shown in Figures 4-16 and 4-17. The unacceptability of the noisy designs at most locations is in sharp contrast with the conformity to noise limits of the quiet designs. Figure 4-18 summarizes the study of DOC sensitivity to external noise restraint. As noted previously, the baseline helicopter achieves the 95 PNdB noise limit goal with rotor parameters selected to minimize DOC, i.e., any change in rotor tipspeed, disc loading, blade loading, etc., whether to increase or decrease noise, tends to increase DOC in the manner shown. For the compound, because rotor design to achieve the noise limit goal cannot be optimum (minimum DOC) the trend of DOC versus noise in Figure 4-18 continues to decrease with increasing noise level, out to the NC design point which does represent rotor parameters selected to minimize DOC.

As suggested above, a variable twist rotor blade would significantly aid in attaining compatibility between rotor hover performance plus low noise requirements and low drag at high speed. Figure 4-19 compares a compound employing such a concept with a compound having fixed geometry blades.

Main and tail rotor parameters for the two families of related design points are compared in Figure 4-20. Figure 4-21 gives summary weight statements for the six aircraft studied. Figures 4-22 through 4-25 are three-view drawings of the QH, NH, QC, and NC designs.
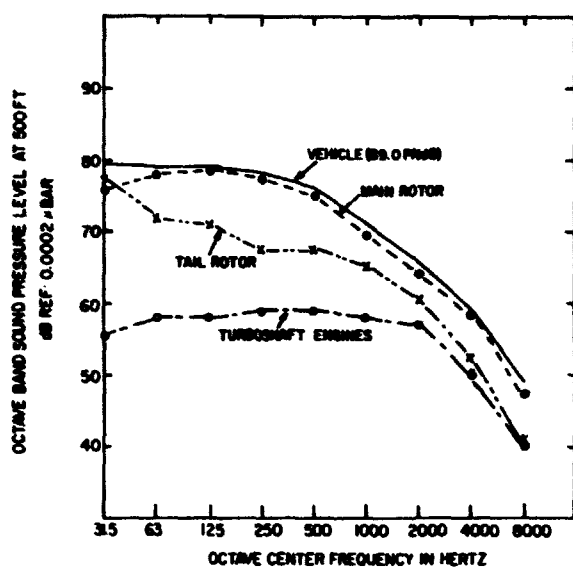
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FIGURE 4-3.   QUIET HELICOPTER
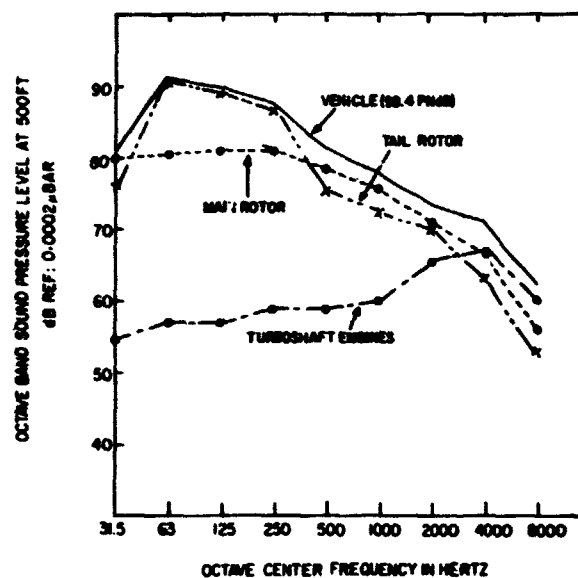EXTERNAL NOISE SPECTRUM
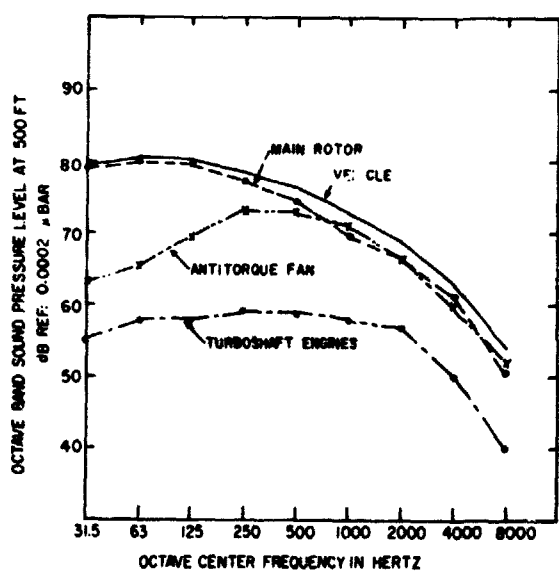

FIGURE 4-4.   NOISY HELICOPTER
EXTERNAL NOISE SPECTRUM
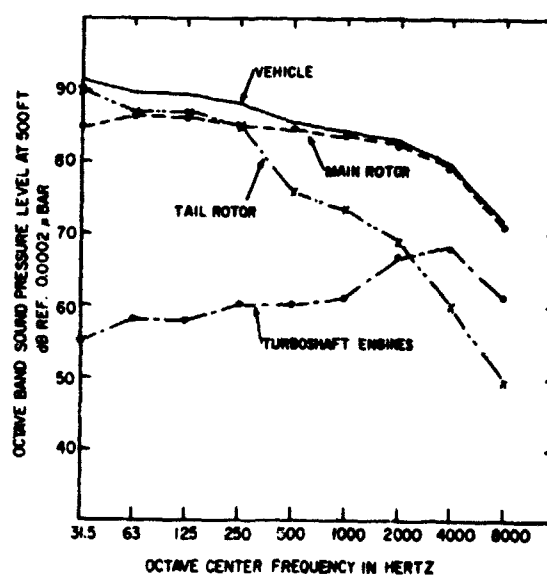

FIGURE 4-5.   QUIET COMPOUND
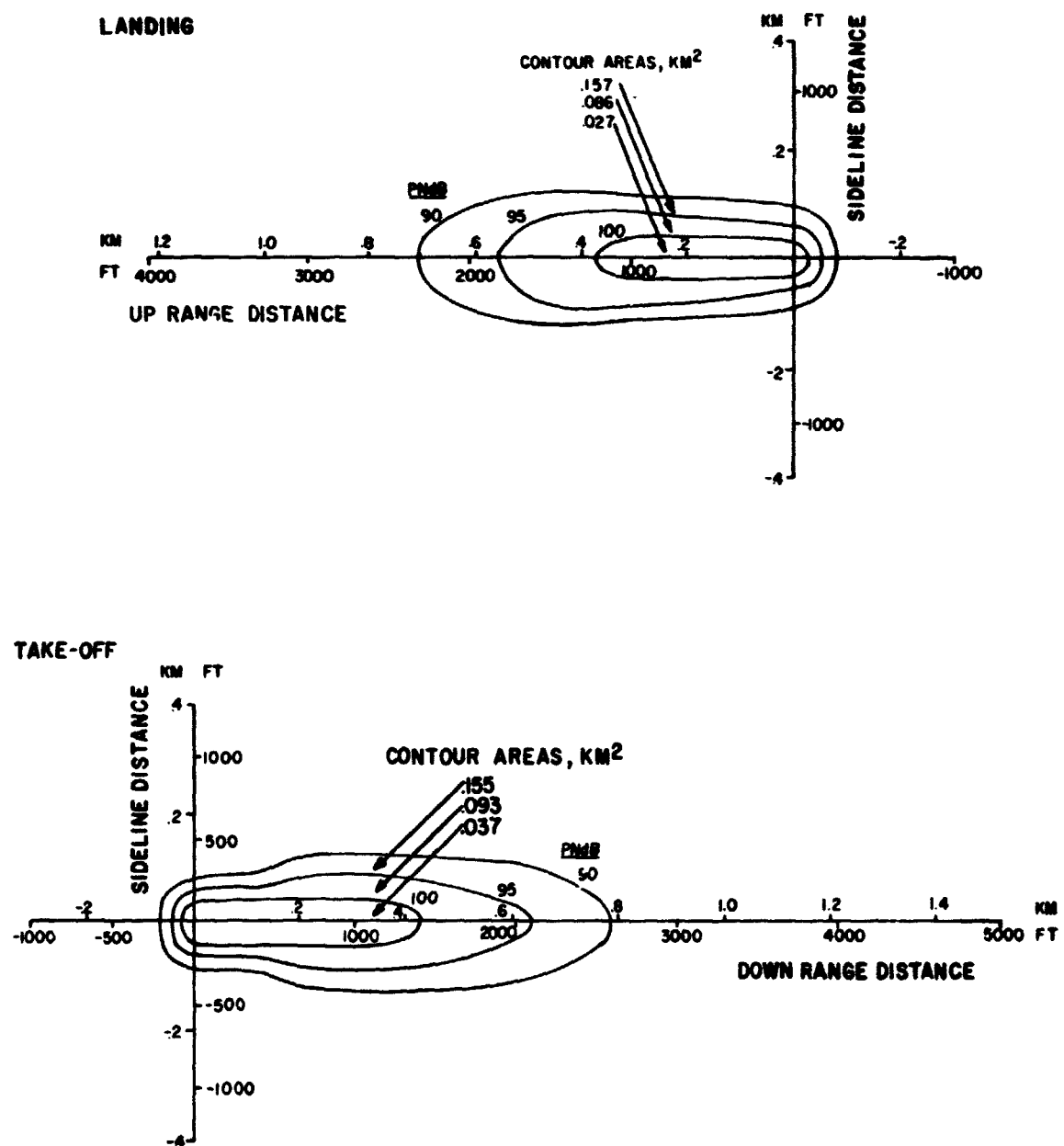EXTERNAL NOISE SPECTRUM

FIGURE 4-6.   NOISY COMPOUND
EXTERNAL NOISE SPECTRUM

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FIGURE 4-7. QUIET HELICOPTER PNL CONTOURS
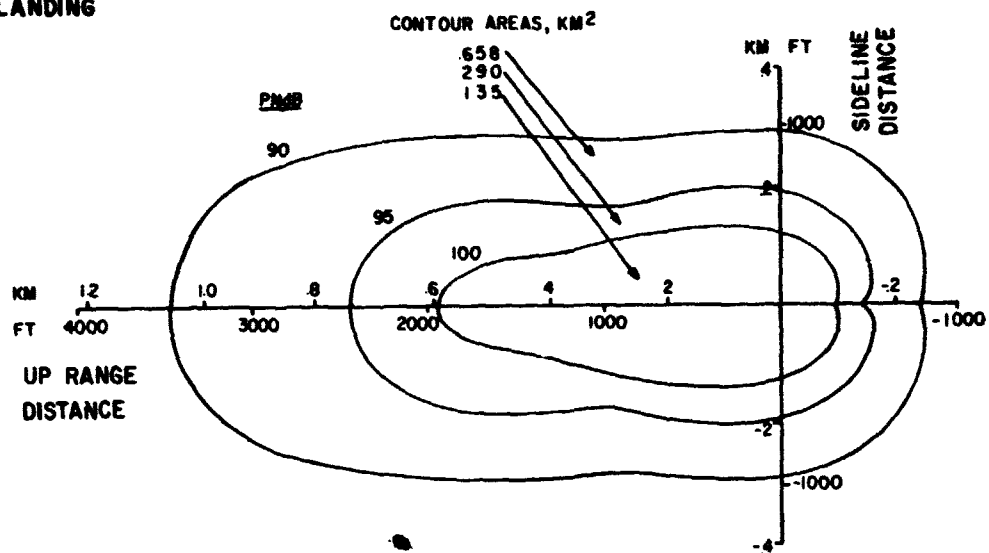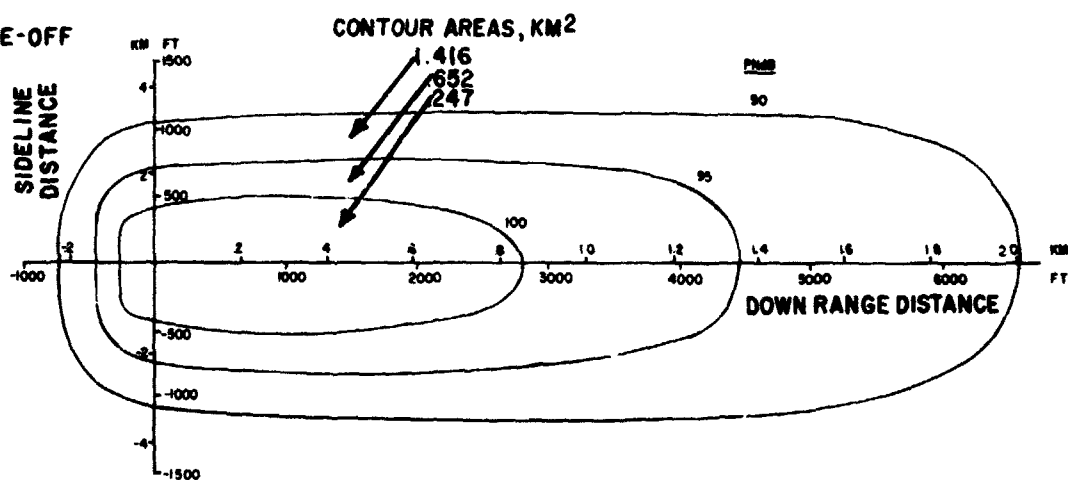
FIGURE 4-8. NOISY HELICOPTER PNL CONTOURS

89
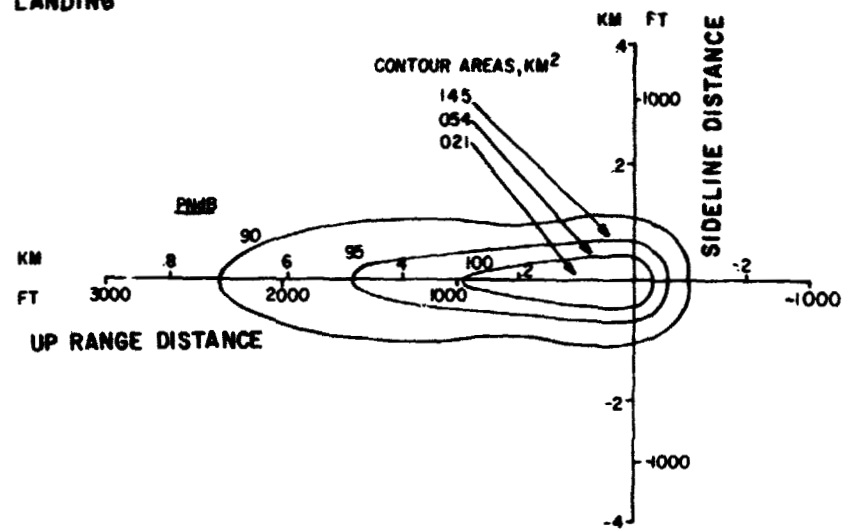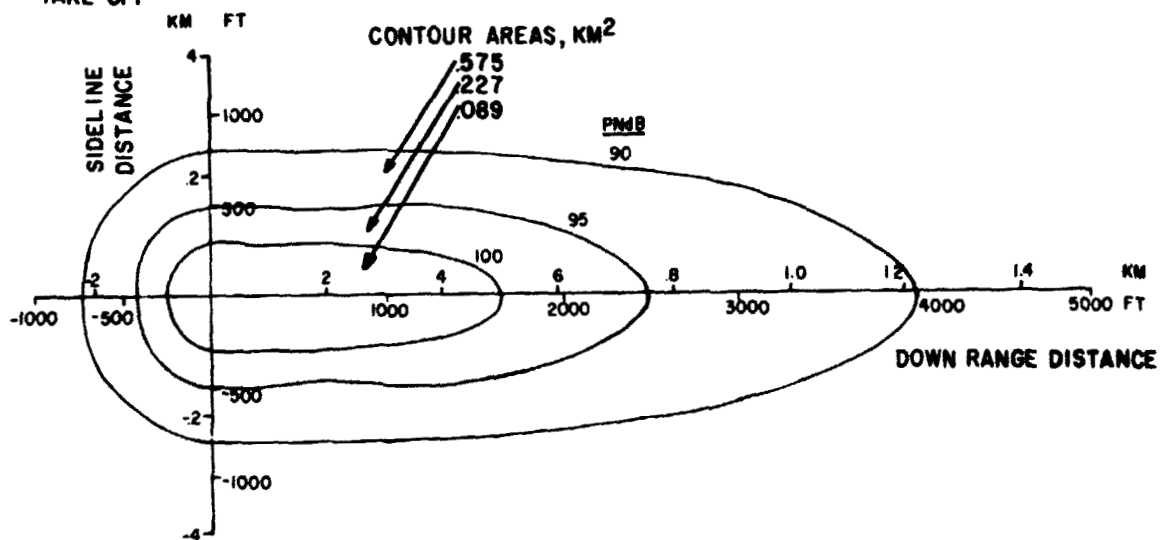
FIGURE 4-9. QUIET COMPOUND PNL CONTOURS
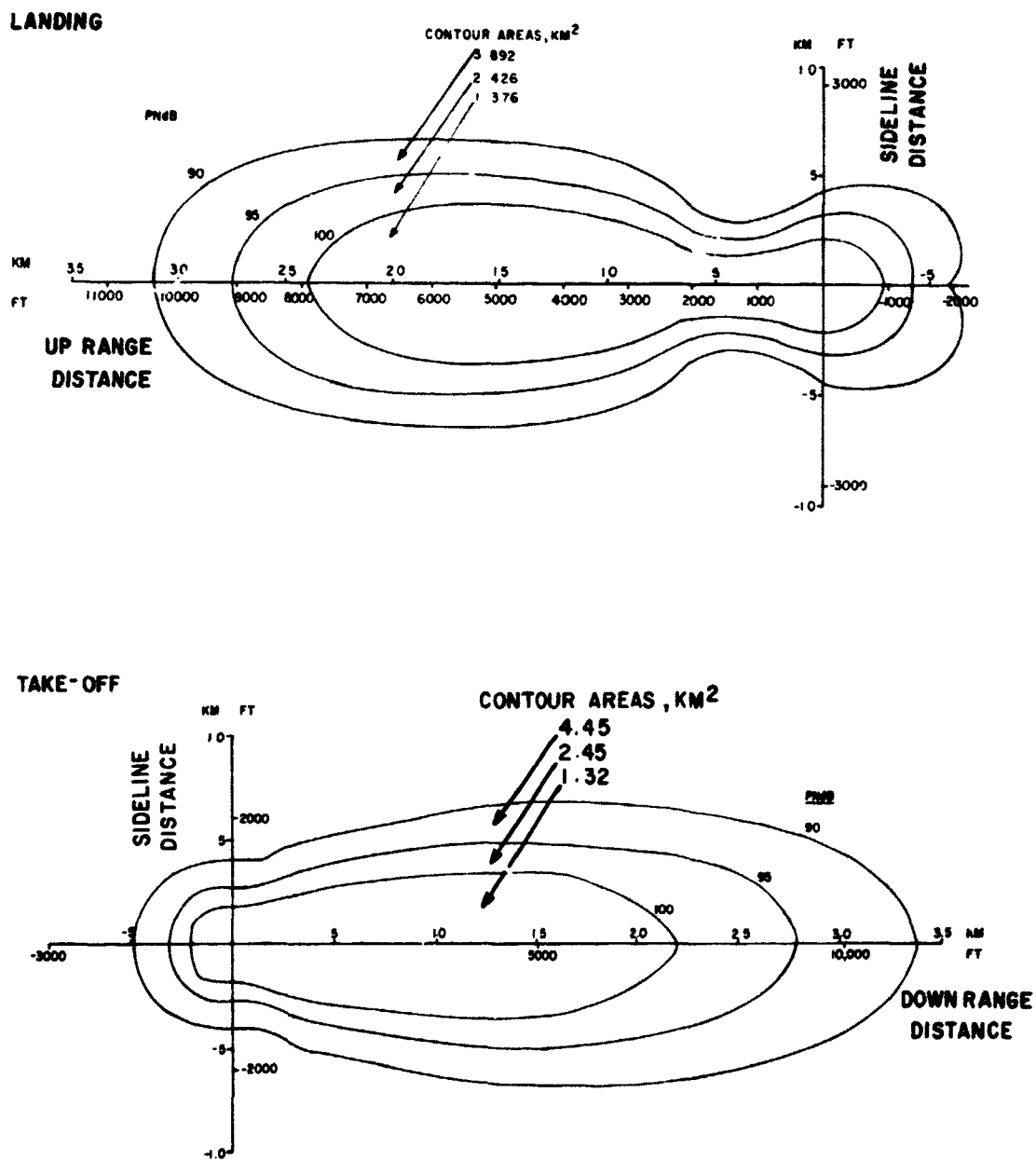
FIGURE 4-10. NOISY COMPOUND PNL CONTOURS

91
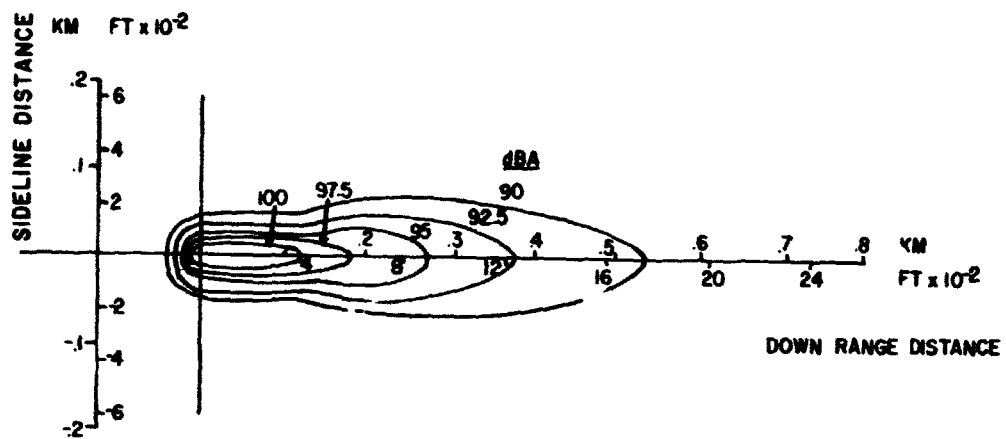
FIGURE 4-11.   QUIET HELICOPTER TAKE-OFF SENEL CONTOURS
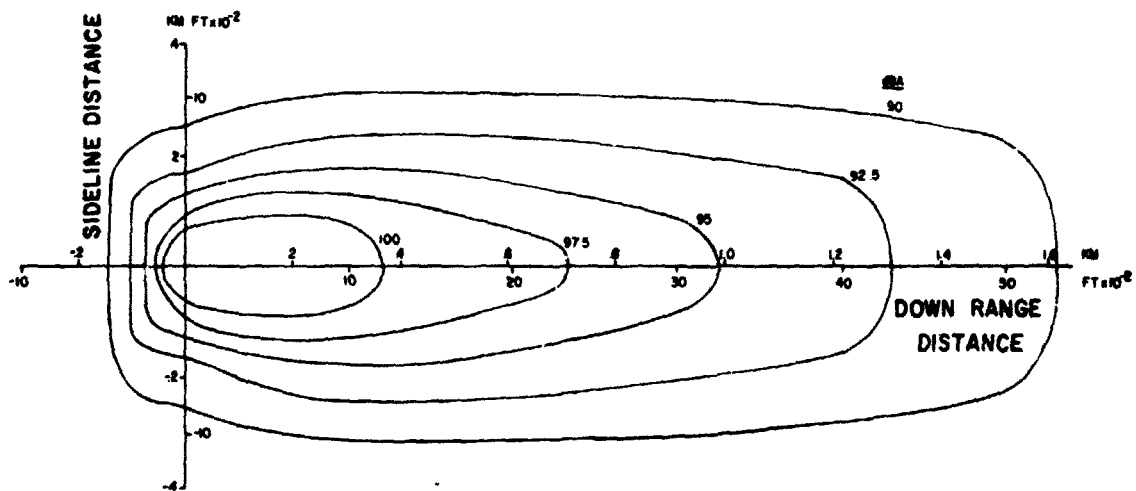
FIGURE 4-12.   NOISY HELICOPTER TAKE-OFF SENEL CONTOURS
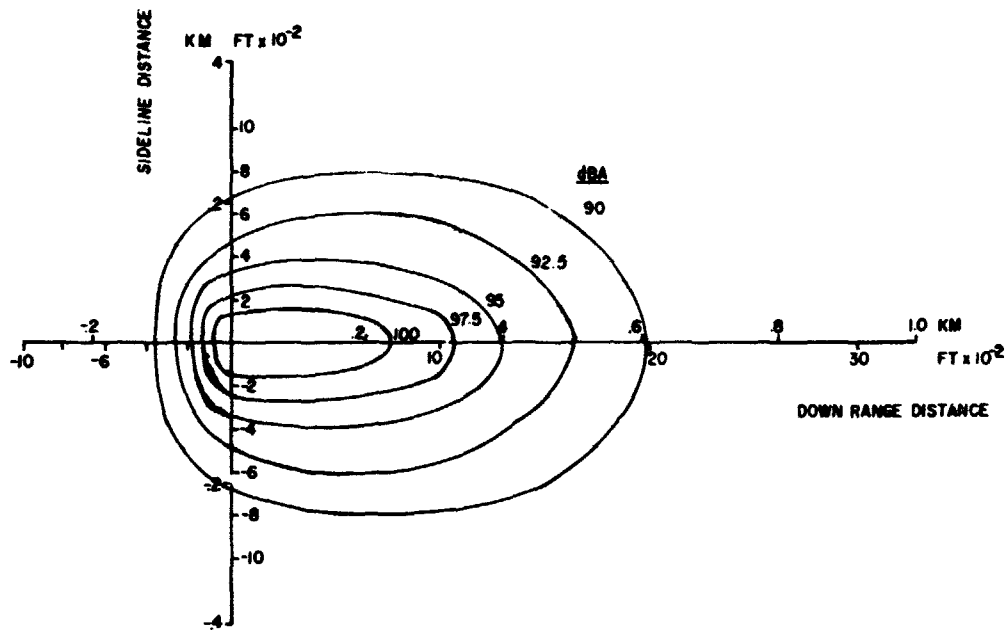
FIGURE 4-13.   QUIET COMPOUND TAKE-OFF SENEL CONTOURS

FIGURE 4-14.   NOISY COMPOUND TAKE-OFF SENEL CONTOURS

FIGURE 4-15. STUDY AIRCRAFT EXTERNAL NOISE vs. CONTOUR AREA

FIGURE 4-16. HELICOPTER SENEL CONTOUR AREA vs. COMMUNITY ACCEPTANCE
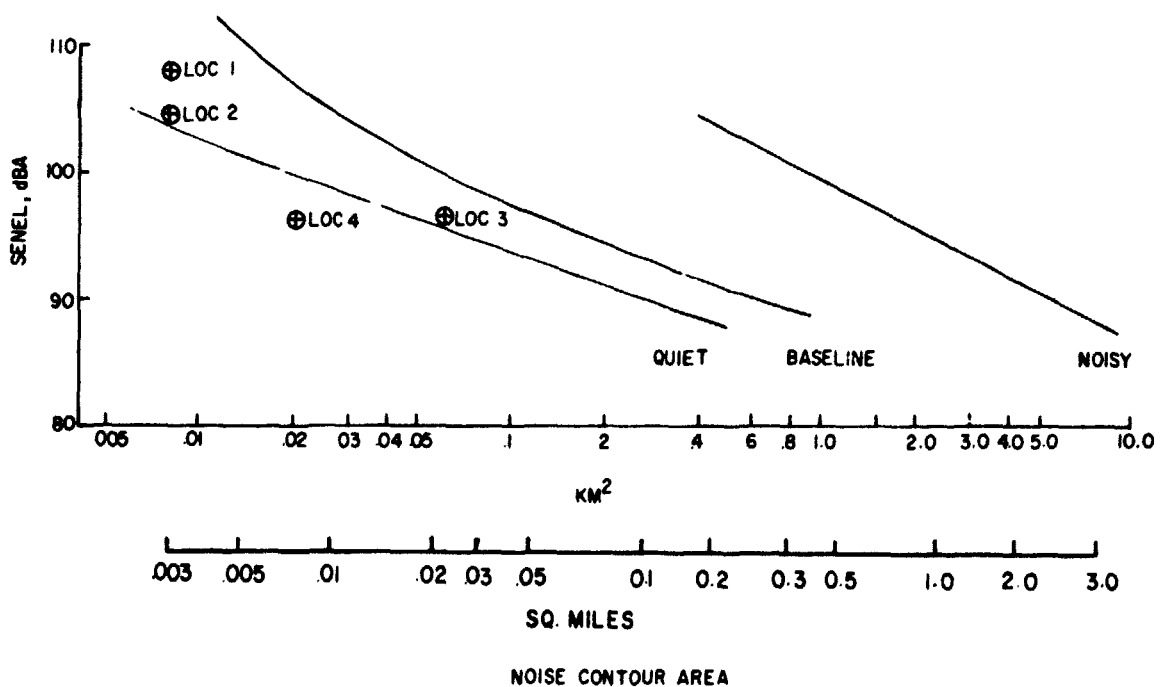
FIGURE 4-17. COMPOUNL SENEL CONTOUR AREA vs. COMMUNITY ACCEPTANCE
CRITERIA

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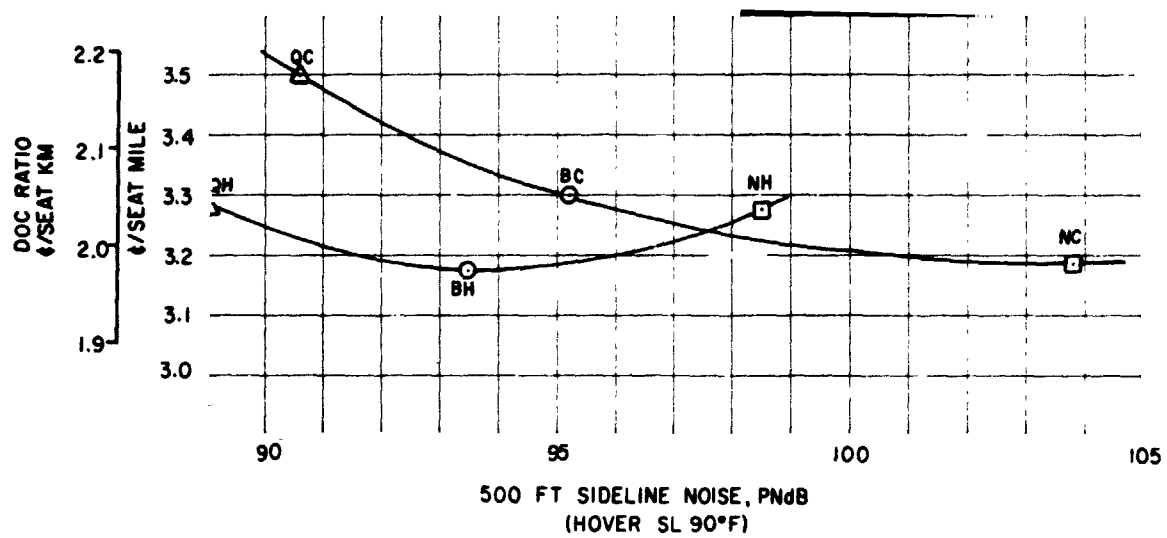
FIGURE 4-18.   DIRECT OPERATING COST vs. EXTERNAL NOISE RESTRAINT
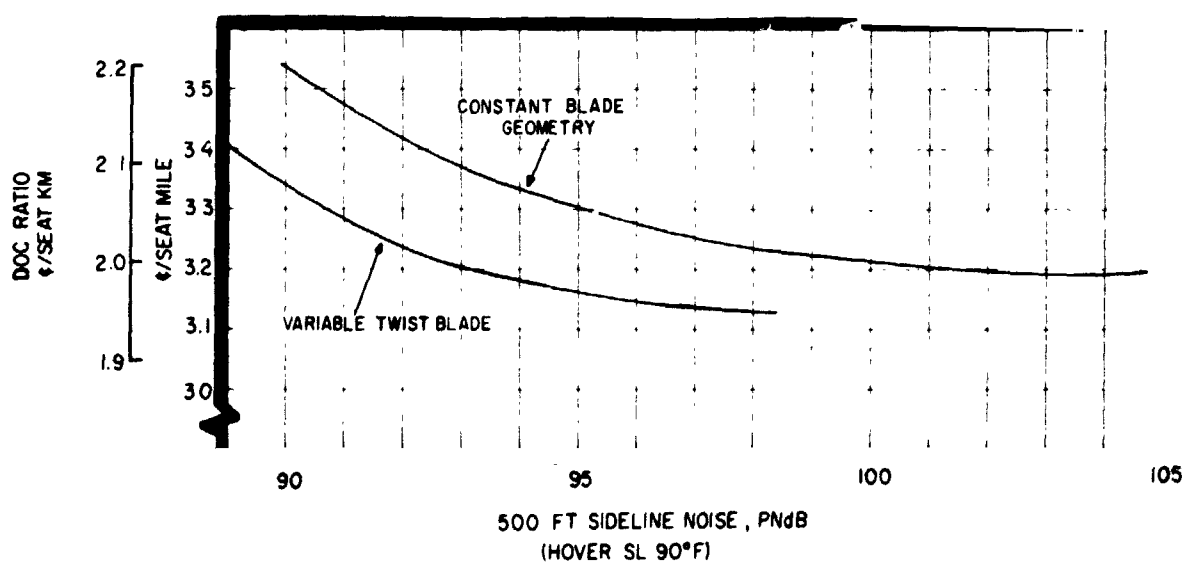
FIGURE 4-19.   COMPOUND - EFFECT OF VARIABLE TWIST BLADE ON DOC/NOISE
RELATIONSHIP

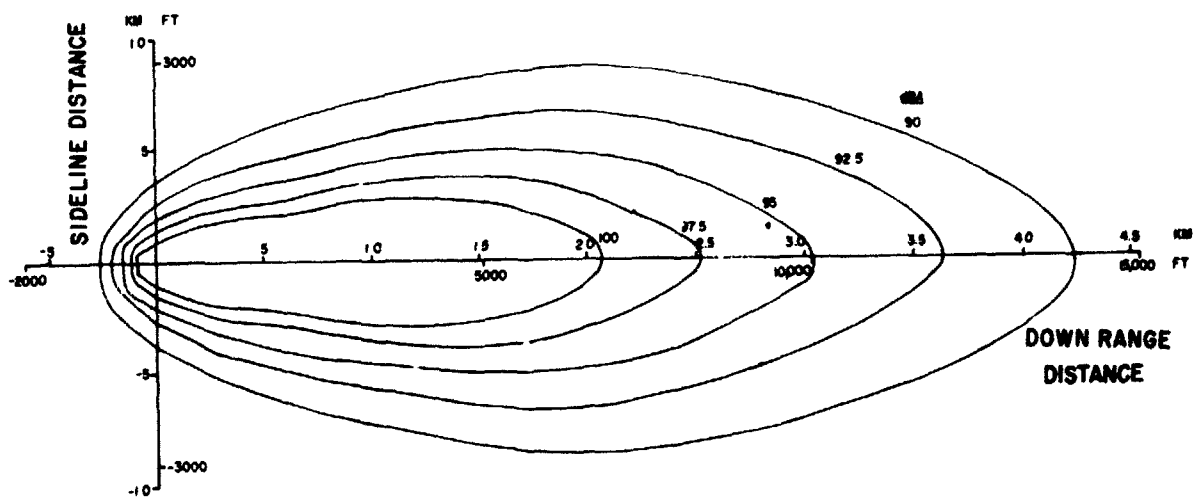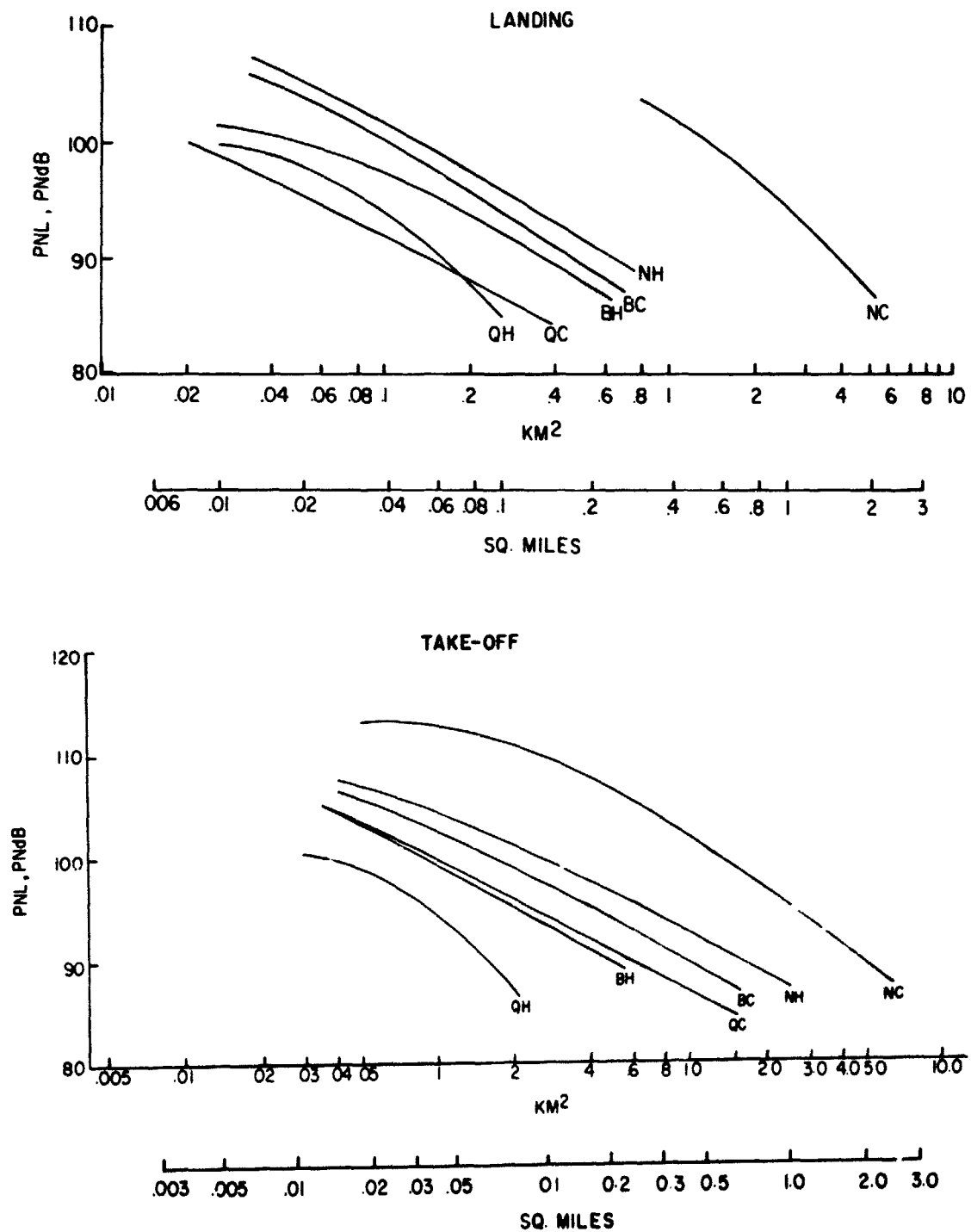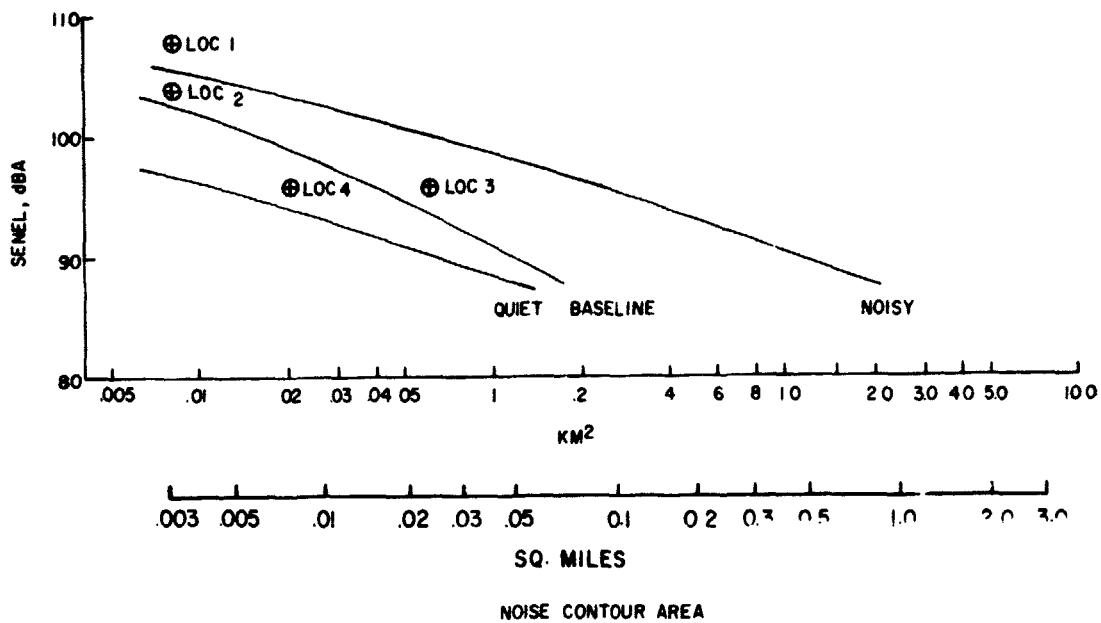|	HELICOPTER			COMPOUND		
	QH	BH	NH	QC	BC	NC
GROSS WEIGHT, kg (lb)	27335 (60269)	26368 (58137)	26584 (58612)	36433 (80327)	34437 (75926)	33246 (73301)
WEIGHT EMPTY, kg (lb)	16643 (36694)	15591 (34374)	15571 (34771)	24189 (53332)	22480 (49564)	21567 (47554)
MAIN ROTOR						
DIAMETER, m (ft)	28.6 (93.9)	28.1 (92.2)	28.2 (92.6)	27 ∟ (90.6)	26.9 (88.4)	26.4 (86.7)
BLADES	7	6	6	7	6	ε
TWIST, DEG	-16	-16	-10	-12	-12	-4
HOVER TIPSPEED, mps (fps)	201 (660)	222.5 (730)	232 (760)	186 (610)	210 (690)	222.5 (730)
CRUISE TIPSPEED, mps (fps)	201 (660)	222.5 (730)	213 (700)	158 (520)	158 (520)	158 (520)
C_T/σ, HOVER @ SL 90°F	.075	.075	.07	.1	.1	.115
ANTI-TORQUE DEVICE						
TYPE	ROTOR (2)	ROTOR	ROTOR*	FAN	ROTOR	ROTOR
DIAMETER, m (ft)	6.64 (21.8)	5.55 (18.2)	5.49 (18.0)	3.14 (10.3)	7.22 (23.7)	6.95 (22.8)
BLADES	4 (each)	6	4	13	6	4
TWIST, DEG	-16	-16	-10	-16	-12	-4
HOVER TIPSPEED, mps (fps)	168 (550)	213 (700)	232 (760)	213 (700)	210 (690)	213 (700)
500 FT SIDELINE NOISE, PNdB	89.0	93.5	98.5	90.6	95.2	103.8
DIRECT OPERATING COST, ¢/SK (¢/SM)	2.037 (3.277)	1.923 (3.174)	2.040 (3.283)	2.175 (3.499)	2.050 (3.299)	1.980 (3.18<)
INITIAL COST, $ X 10^6	4.126	3.948	3.832	6.032	5.673	5.418

* Main-tail rotor clearance reduced to 15.24 cm (6 inches).

FIGURE 4-20. COMPARISON OF RELATED 100-PASSENGER DESIGNS

COMPONENT WEIGHTS (KG, LB)

	QH		BH		NH		QC		BC		NC	
	KG	LB	KG	LB	KG	LB	KG	LB	KG	LB	KG	LB
MAIN ROTOR GROUP	2574	5675	2313	5099	2507	5528	2880	6349	2434	5367	1864	4109
WING GROUP	0	0	0	0	0	0	954	2103	896	1975	855	1886
TAIL GROUP												
TAIL ROTOR/FAN	370	815	168	370	137	303	264	582	356	785	346	763
TAIL SURFACES	402	886	377	832	377	832	537	1184	473	1043	473	1043
BODY GROUP	3137	6917	2988	6587	2996	6606	3747	8260	3666	8082	3608	7954
ALIGHTING GEAR	670	1477	651	1435	647	1427	843	1859	806	1777	783	1727
FLIGHT CONTROLS	629	1386	609	1343	614	1353	855	1884	827	1824	735	1621
ENGINE SECTION	255	563	237	523	245	540	665	1465	577	1273	638	1235
PROPULSION GROUP												
ENGINES	974	2147	916	2020	941	2075	1826	4025	1526	3365	1388	3059
AIR INDUCTION	35	78	33	72	34	74	20	44	16	36	15	32
EXHAUST SYSTEM	22	48	20	45	21	47	42	92	34	74	30	66
LUBRICATING SYSTEM	0	0	0	0	0	0	0	0	0	0	0	0
FUEL SYSTEM	158	348	164	362	167	368	122	260	112	246	102	226
ENGINE CONTROLS	35	78	33	72	34	74	71	157	59	129	53	116
STARTING SYSTEM	93	204	86	189	88	195	189	416	155	341	139	307
AUXILIARY PROPULSION PROPELLERS	0	0	0	0	0	0	2111	4654	2055	4533	1997	4402
DRIVE SYSTEM	2815	6207	2519	5553	2506	5524	3816	8412	3292	7251	3500	7717
AUXILIARY POWER UNIT	266	586	266	586	266	586	266	586	266	586	266	586
INSTRUMENTS	262	577	262	577	262	577	293	645	293	645	293	645
HYDRAULICS	70	155	70	155	70	155	78	173	78	173	78	173
ELECTRICAL GROUP	330	728	330	728	330	728	391	862	391	862	391	862
AVIONICS	298	658	298	658	298	658	298	658	298	658	298	658
FURNISHINGS	2495	5502	2505	5523	2475	5456	2712	5981	2747	6055	2739	6039
AIR CONDITIONING AND ANTI-ICE	708	1561	708	1561	708	1561	733	1617	733	1617	733	1617
AUXILIARY GEAR	20	43	20	43	20	43	20	43	20	43	20	43
CONTINGENCY	24	54	19	41	20	44	459	1013	371	818	325	718
WEIGHT EMPTY	16640	36693	15592	34371	15764	34754	24192	53332	22482	49564	21570	47554
FIXED USEFUL LOAD												
PILOT	86	190	86	190	86	190	86	190	86	190	86	190
CO-PILOT	86	190	86	190	86	190	86	190	86	190	86	190
OIL-ENGINE	18	40	18	40	18	40	18	40	18	40	18	40
-TRAPPED	7	16	7	16	7	16	7	16	7	16	7	16
FUEL-TRAPPED	5	11	5	12	5	12	36	79	33	73	30	67
ATTENDANTS	127	280	127	280	127	280	127	280	127	280	127	280
MISSION EQUIPMENT	136	300	136	300	136	300	136	300	136	300	136	300
PAYLOAD	8165	18000	8165	18000	8165	18000	8165	18000	8165	18000	8165	18000
FUEL-USABLE	2062	4548	2149	4735	2184	4813	3583	7900	3300	7273	3024	6664
GROSS WEIGHT	27332	60268	26371	58127	26578	58595	36436	80327	34440	75926	33249	73301

FIGURE 4-21. STUDY AIRCRAFT WEIGHT STATEMENT.

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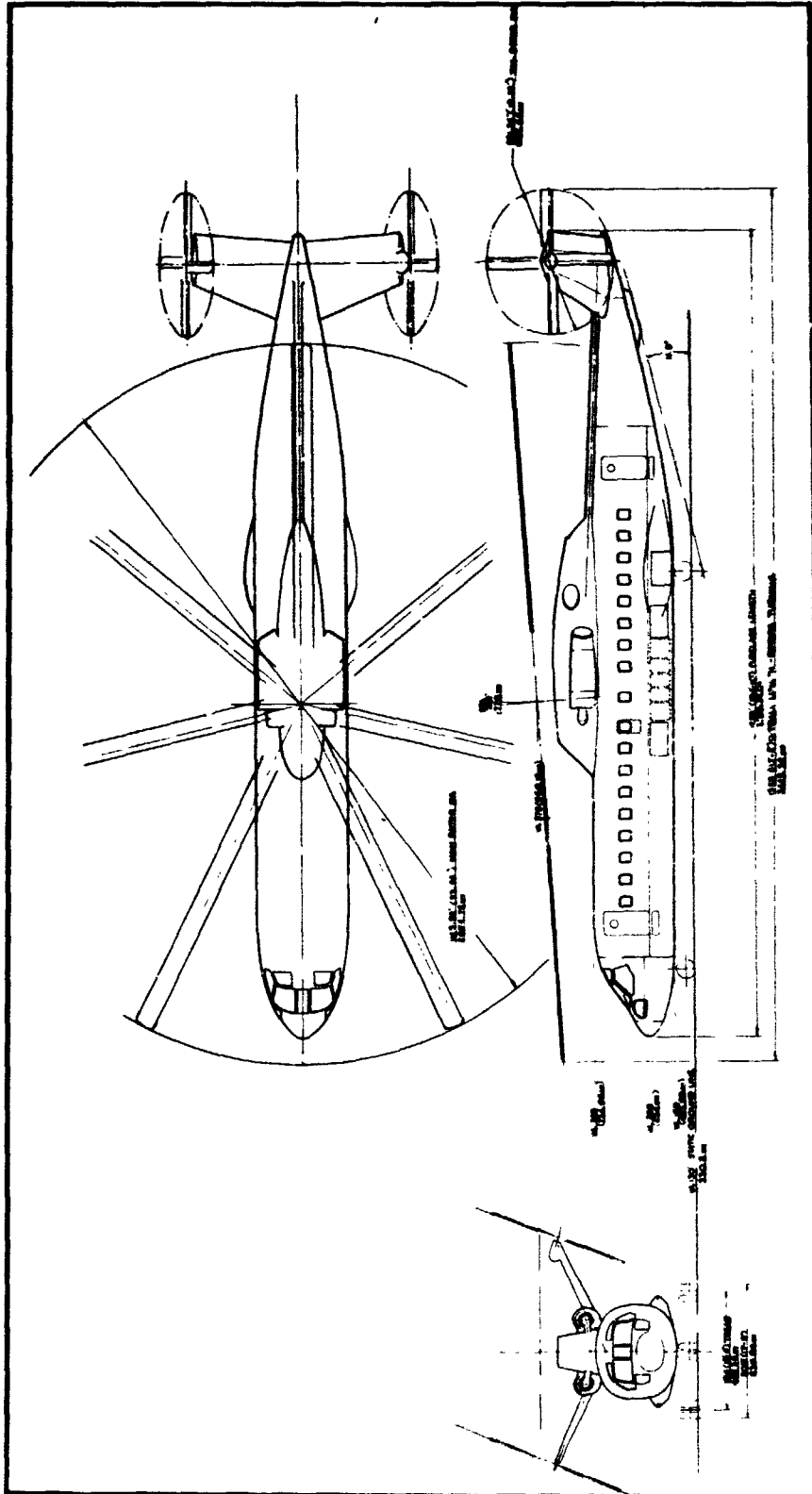
FIGURE 4-22. QUIET HELICOPTER 3-VIEW
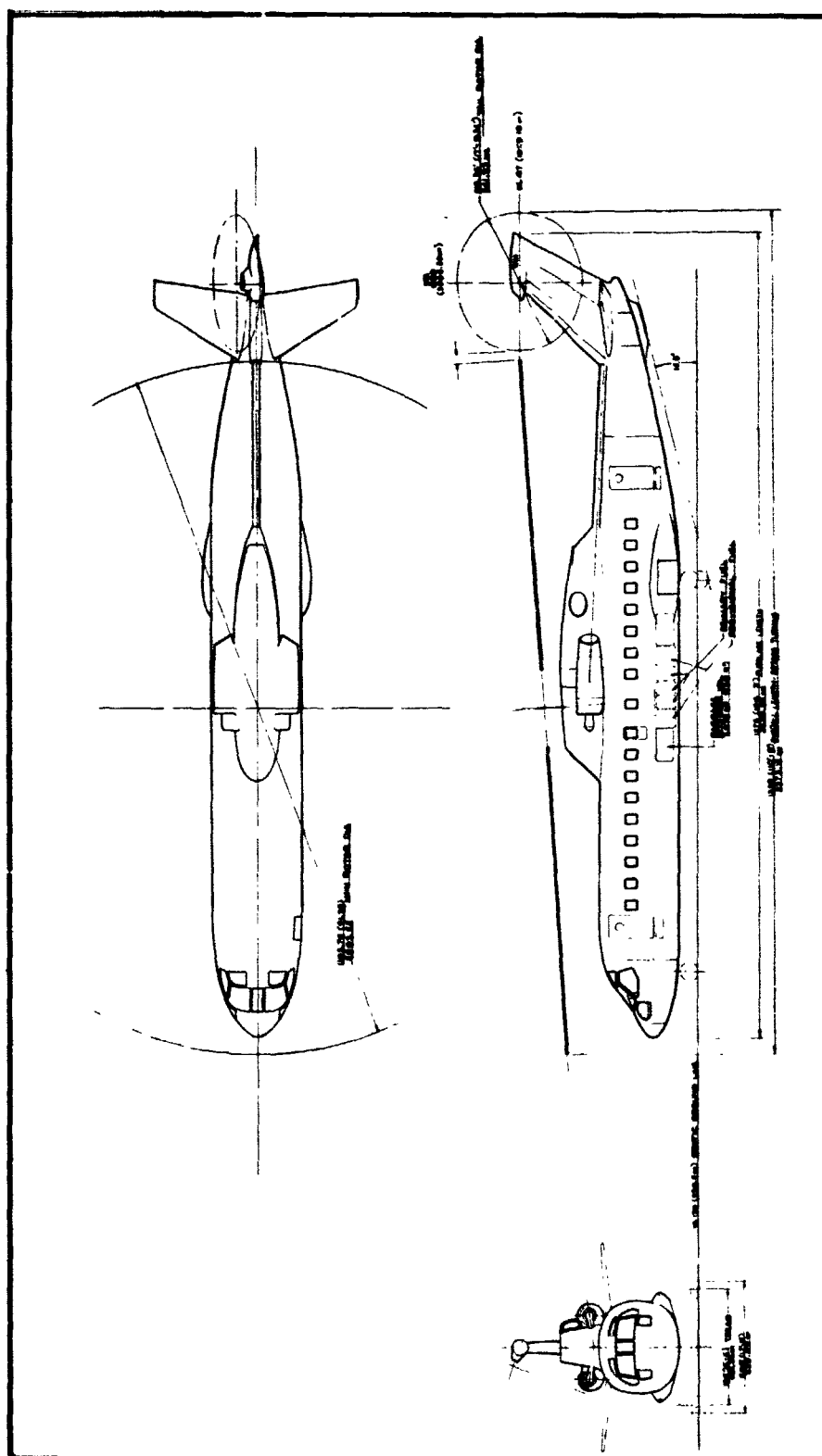
99

FIGURE 4-23. NOISY HELICOPTER 3-VIEW

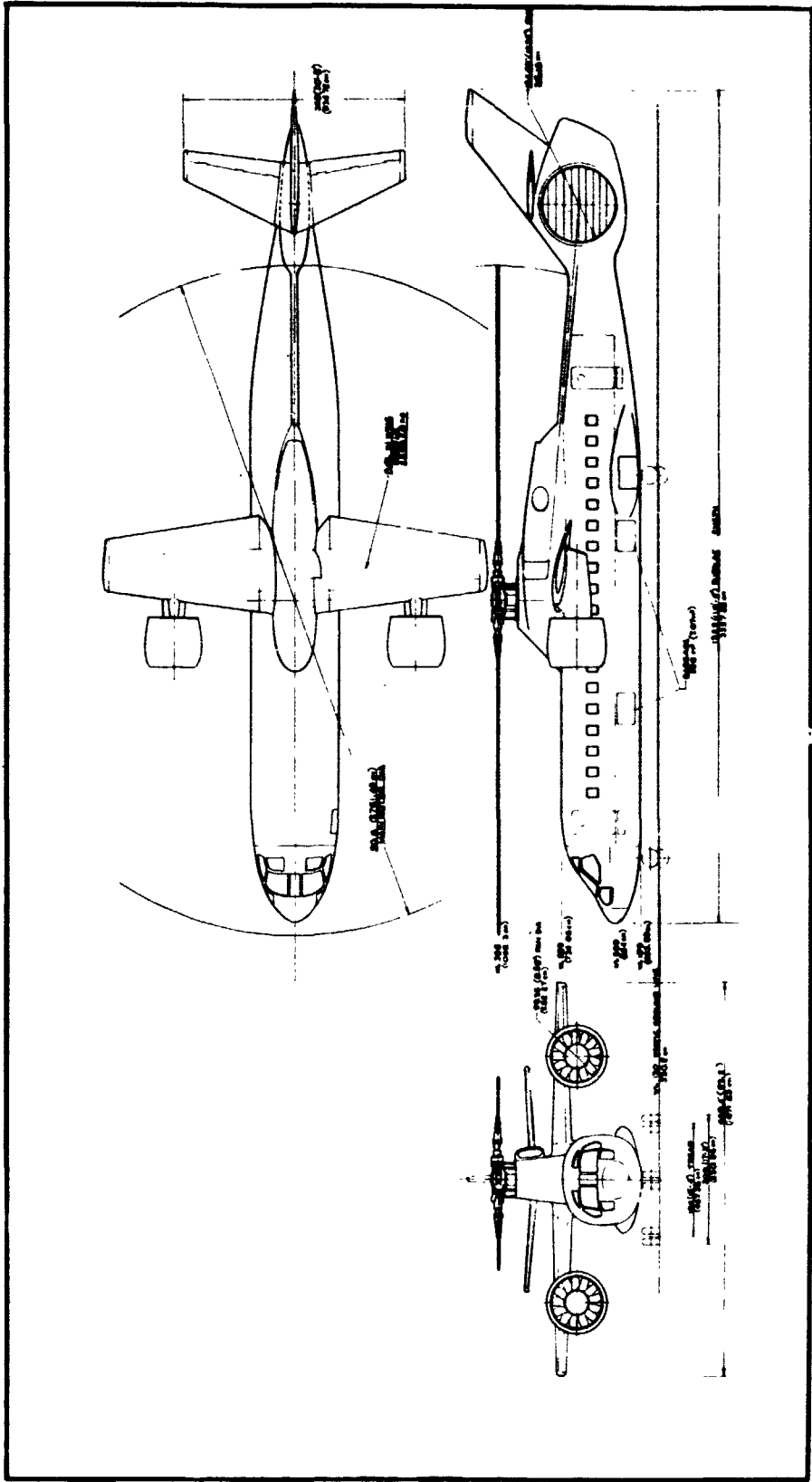FIGURE 4-24. QUIET COMPOUND 3-VIEW

FIGURE 4-25. NOISY COMPOUND 3-VIEW

4.3 Effects on Handling Qualities

The quiet and noisy helicopter and compound aircraft were designed to the same criteria as the baseline helicopter and compound. The same fuselage aerodynamic characteristics were used for the helicopters and compounds. In addition, the compound wing loadings and propulsor efficiencies are the same as those of the baseline compound. The vertical tail was designed so that each aircraft could be flown following a loss of anti-torque thrust. The horizontal tails were designed to provide the same level of static stability achieved in the baseline helicopter and compound. In light of this design approach, the helicopter and compound aircraft can be expected to have trim, stability, and response characteristics similar to those of the baseline aircraft.

The head moment constants of the helicopters and compounds are compared in Figure 4-26. As can be seen, the head moment constants of the helicopters are all nearly equal, so the quiet and noisy helicopters can be expected to have response characteristics similar to those of the baseline with the same control input. The head moment constants of the compound are nearly equal for the baseline and quiet aircraft, so these two should have similar response characteristics. The noisy compound head moment constant is approximately 23% less than the baseline and thus would require 23% more control input about trim point for the same level of maneuverability.

Helicopter	Head Moment Constant M-kg/deg (FT-LB/DEG.)	
Base Line	1266.	(9162)
Quiet	1320.	(9555)
Noisy	1324.	(9578)
Compound	Head Moment Constant M-kg/deg (FT-LB/DEG.)	
Base Line	1461.	(10570)
Quiet	1406.	(10176)
Noisy	1125.	(8140)

FIGURE 4-26.   STUDY AIRCRAFT HEAD MOMENT CONSTANTS

5.0 DOC TRENDING

5.1 Range

Baseline stagelength was 370 kilometers (200 nautical miles), but the
effect on DOC of varying range from 93 to 741 kilometers (50 to 400 nautical
miles) was assessed for the two baseline aircraft. For ranges less than the
baseline, fuel capacity (fuel system weight), was unaltered. For ranges greater
than the baseline, sufficient fuel system weight increase was assessed, and
passengers were off-loaded so that the design gross weight was unaltered.
Figures 5-1 and 5-2 show results for the helicopter and compound, respectively.
As range is increased, the attraction of the compound in reducing trip time is
increased, Figure 5-3, but the higher fuel consumption rate, Figure 5-4, means
that more passengers must be off-loaded. At 741-kilometers range (400 nautical
miles), compound payload has been reduced to 74 passengers, compared to 83 for
the helicopter. Increasing range tends to decrease DOC because the effects of
take-off, climb, descent, and land on block time are reduced. However, the off-
loading of passengers becomes the more powerful effect, so that at 741-kilometers
range, compound DOC has increased by 18% over the baseline value, the helicopter
by 9%. At short ranges, where high speed is not rewarded in terms of DOC, the
helicopter is clearly more economical to operate.

5.2 Utilization

Figure 5-5 shows the effect on DOC of varying aircraft utilization, for
both aircraft. Twenty-five hundred hours a year, considered the most meaningful
for this size and class of aircraft, was used to define the base DOC. DOC can
be reduced by about 9% when utilization is increased to 3500 hours per year. It
may be significant that the AIA cost formula considers utilization as a function
of block time (Reference 3, Figure 1). This trend line indicates a utilization
of 3300 hours for the helicopter and 3000 hours for the compound. This suggests
a 2.7% DOC relative advantage for the slower helicopter that is not evident in
the baseline studies at fixed utilization.

5.3 Manufacturing Cost

Figures 5-6 and 5-7 show the effects on DOC of varying airframe and dynamic
system manufacturing costs. Base assumptions were $243/kg ($110/lb) and $176/kg
($80/lb) respectively.

5.4 Fuel Cost

Figures 5-8 and 5-9 show the effects on DOC of 100% and 200% increases in
fuel cost when each baseline aircraft is operated at a cruise speed within its
design capability. For the helicopter, it is seen that reduced speed always
degrades DOC, whatever the fuel cost in the range considered. For the compound,
if fuel cost were to rise by more than 100%, DOC would be improved by slowing
the aircraft from 129 m/sec (250 knots) to 118 m/sec (230 knots). The base
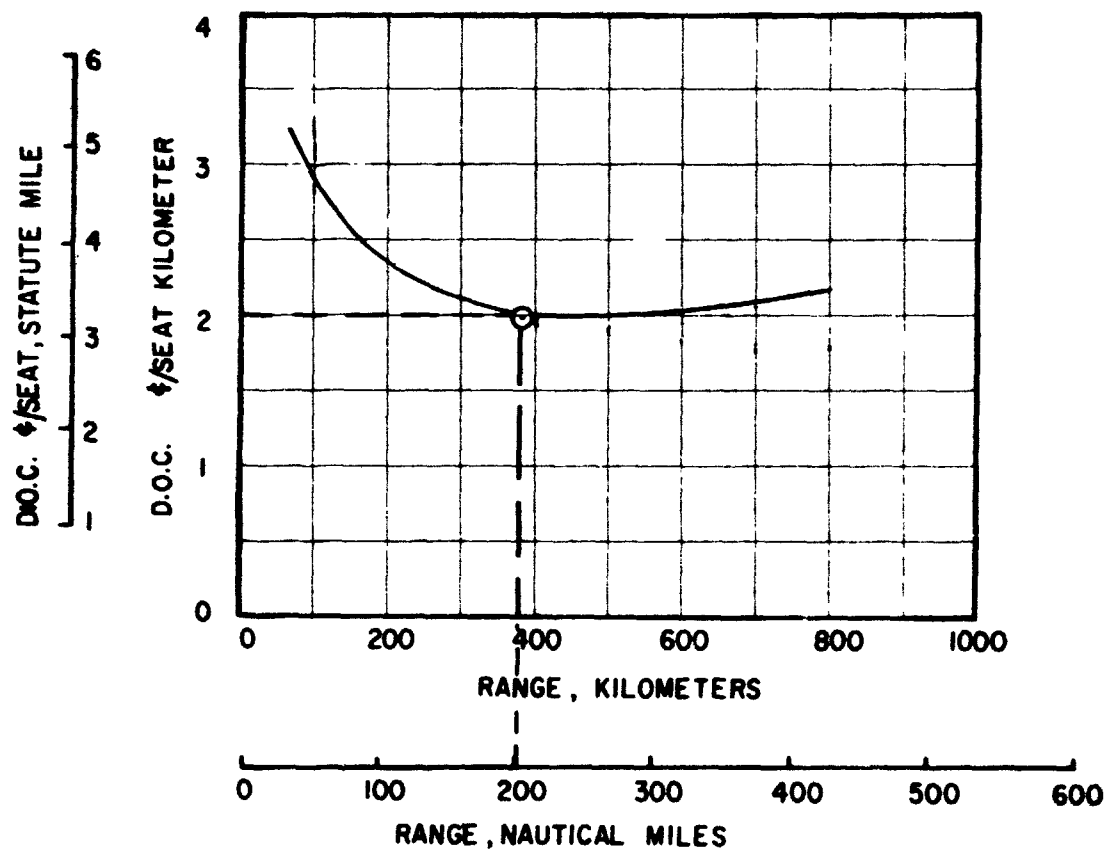fuel cost is 3.43 cents per liter (13 cents per gallon).
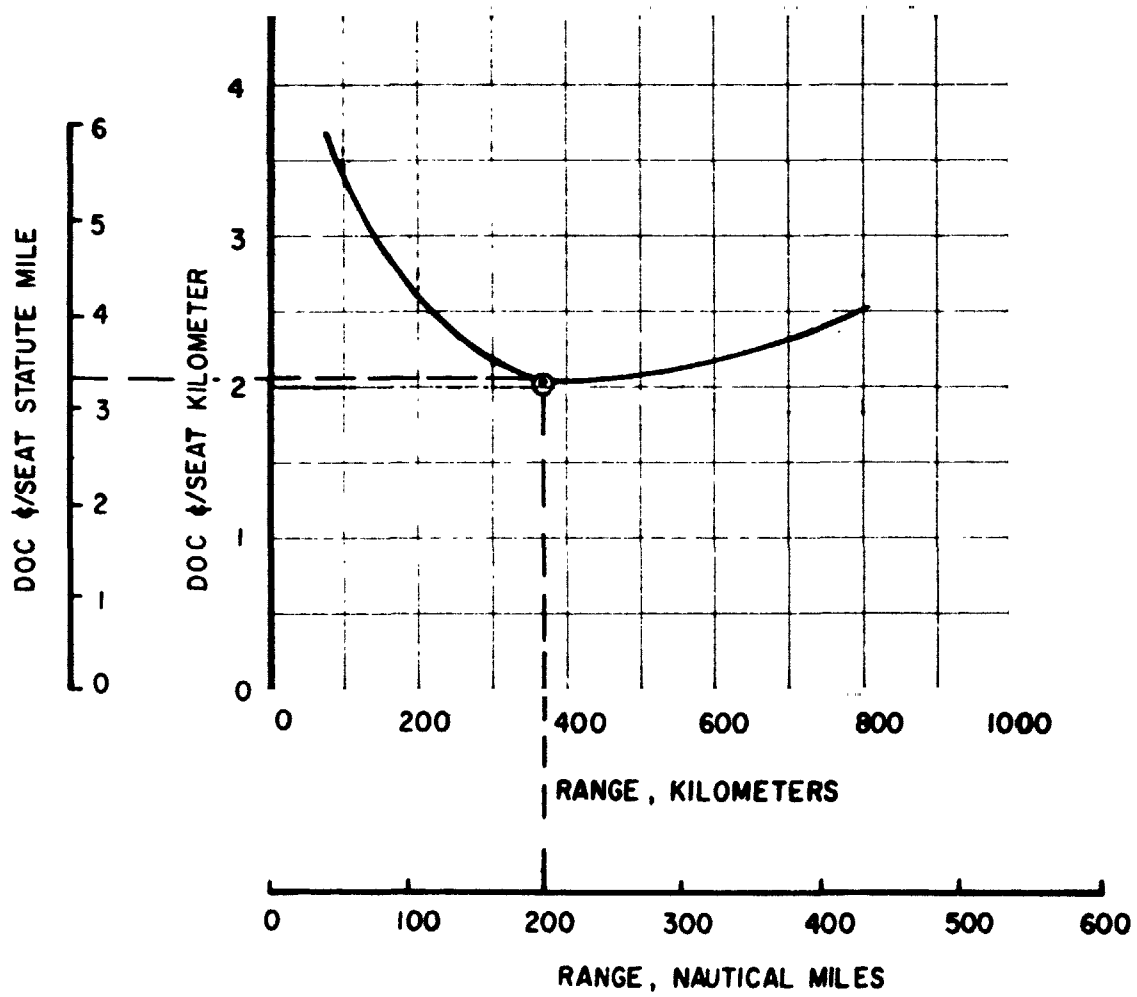
FIGURE 5-1. HELICOPTER DOC vs. RANGE

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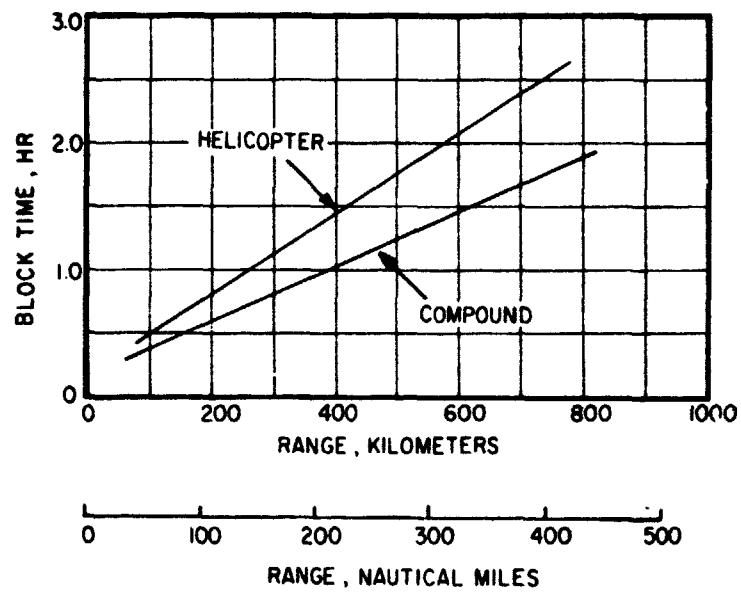FIGURE 5-2.   COMPOUND DOC vs. RANGE

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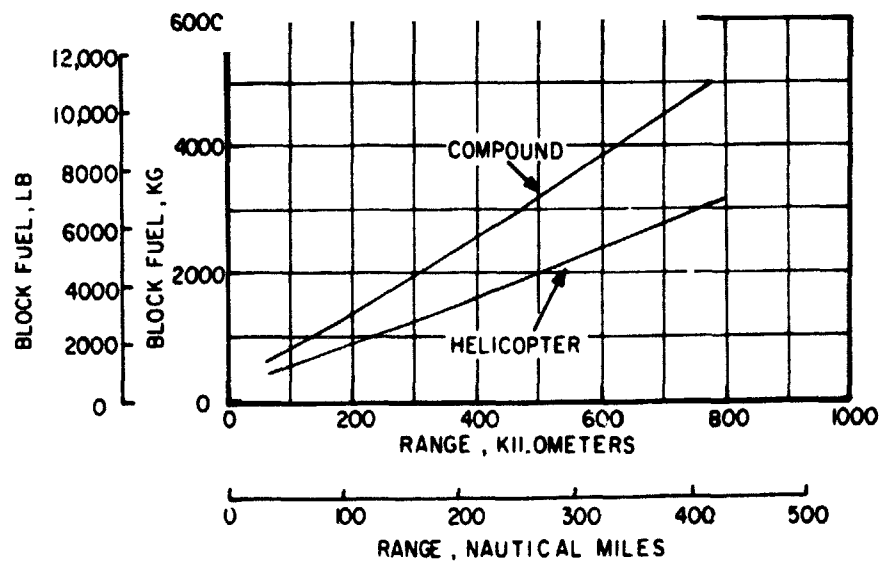FIGURE 5-3. BLOCK TIME vs. RANGE
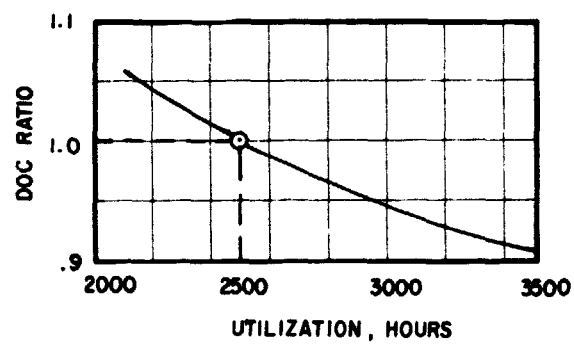

FIGURE 5-4. BLOCK FUEL vs. RANGE
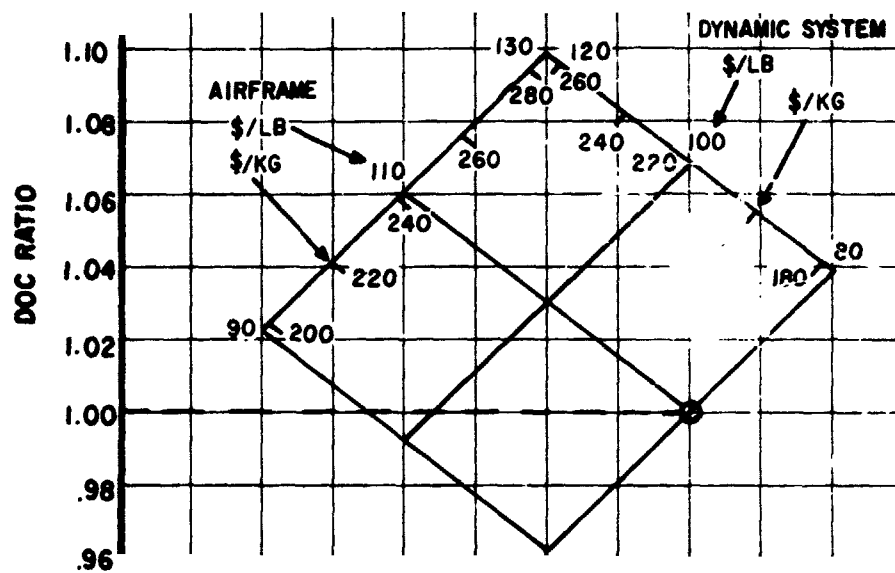
FIGURE 5-5. HELICOPTER AND COMPOUND DOC vs. UTILIZATION
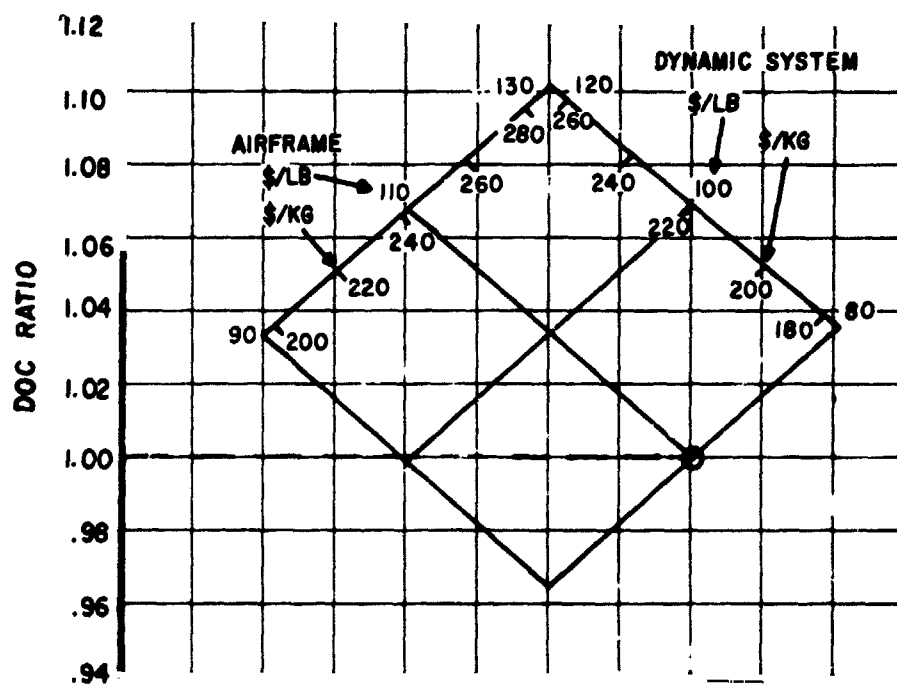
FIGURE 5-6. HELICOPTER DOC vs. MANUFACTURING COST

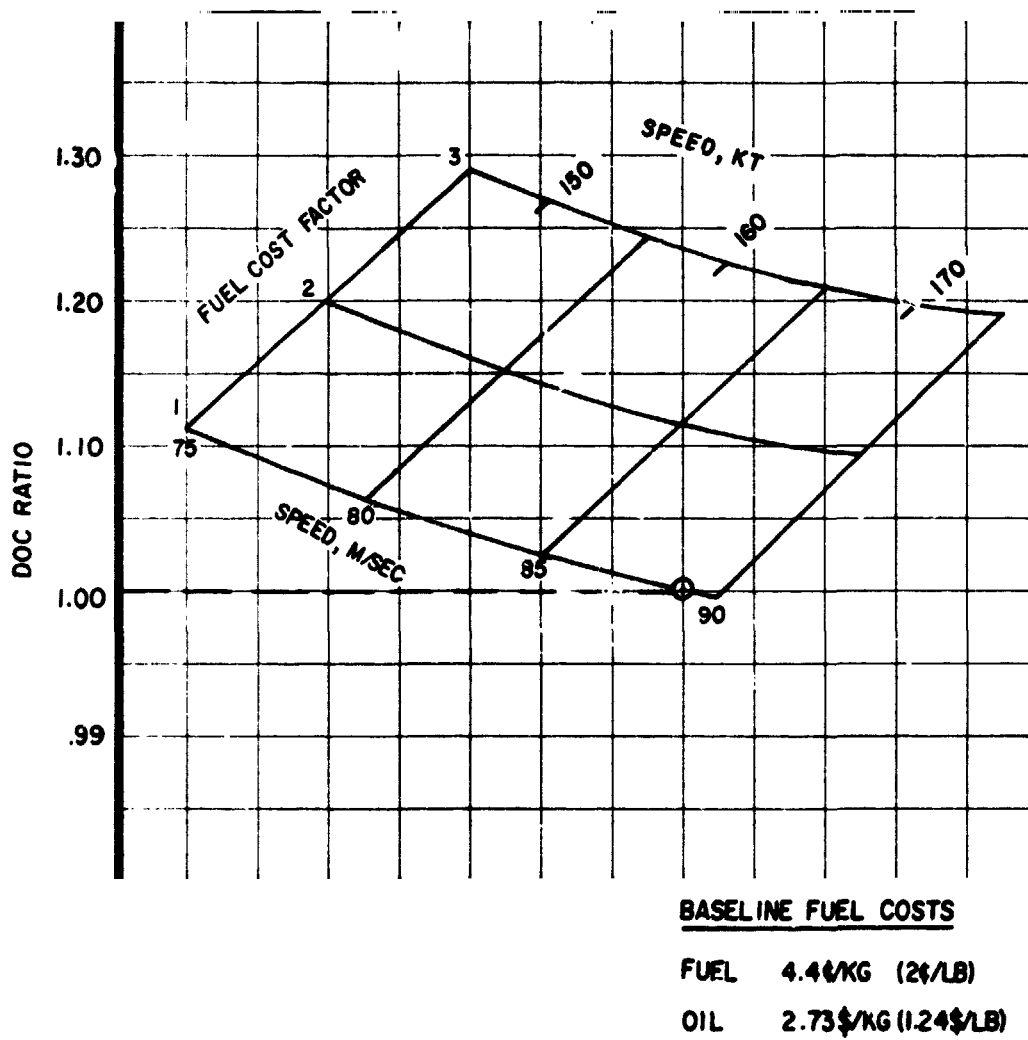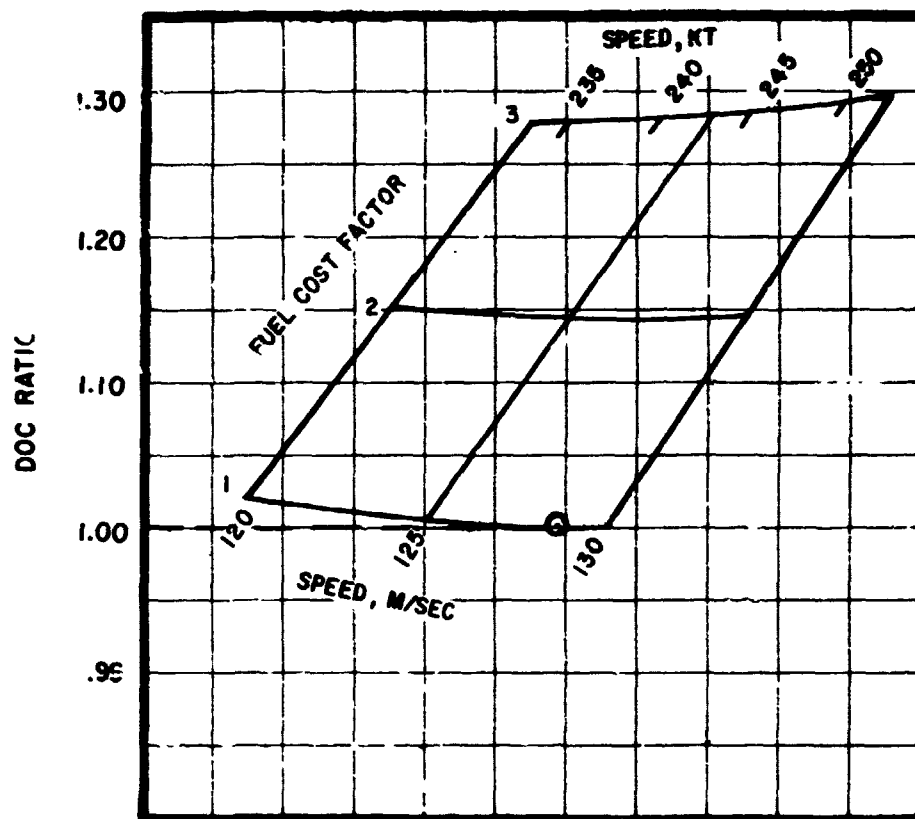

FIGURE 5-7. COMPOUND DOC vs. MANUFACTURING COST

FIGURE 5-8. HELICOPTER DOC vs. SPEED AND FUEL COST

FIGURE 5-9. COMPOUND DOC vs. SPEED AND FUEL COST

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6.0 TECHNICAL RISK ASSESSMENT

The technical risk of scaling up to the 100 passenger helicopter and compound is considered small, as discussed in Section 2.4.1.

The following design features are considered technical risk items:

. <u>Transmission Isolation/Engine Interface</u>. To limit cabin sound and vibration levels to desired levels, the main gearbox must be isolated from the airframe by means of absorptive mounts. The transmission/engine interface must, therefore, be designed to tolerate small relative deflections. For the compound, with two of its three engines wing-mounted, the problem is less acute. Large amounts of main rotor power are experienced only in hover and low speed flight, and only one engine is short-coupled to the main gearbox. The RSRA, for which a transmission isolator has been designed, is expected to develop proper design techniques to overcome this potential problem.

. <u>Fly-By-Wire Control System</u>. Although an innovation for any current production helicopter, fly-by-wire control systems are flying in experimental fixed wing aircraft. The RSRA will have a fly-by-wire system for the pilot's stick, so proper techniques will be learned for mechanical/electrical/mechanical interfacing.

. <u>Convertible Propulsion System</u>. For the compound, the two wing-mounted engines can provide straight-through shaft power to the fan propulsors and/or power to the main gearbox by means of take-off drive shafts running to the main gearbox. A two-speed input section to the main gearbox provides reduced rotor tipspeed in cruise flight and eliminates the need to de-clutch the fans in hover. Although all elements of this propulsion system are proven, some technical risk is associated with integration of the system.

. <u>Twin Tail Rotors (Quiet Helicopter)</u>. To reduce the component of external noise produced by the tail rotor, disc loading must be reduced. Because an unacceptably large, single tail rotor would result, twin devices of low disc loading and providing half of the required thrust are required for the QH design. Technical risk reflects the uncertain knowledge of the mutual flow interference effects on performance, vibration, and noise signature.

7.0 CONCLUSIONS

1. A 100-passenger commercial helicopter can be designed for initial fabrication in 1980. It conforms to a 95 PNdB external noise criterion at 150 meter (500 foot) sideline distance with no compromise to a rotor system chosen specifically to minimize DOC.

2. A 100-passenger compound is compromised in order to achieve the external noise goal, in that blade twist must be increased, main rotor tipspeed and hover blade loading must be decreased significantly from values selected specifically to minimize DOC.

3. Helicopter DOC is 4% lower than that for the compound designed to the same external noise criterion. If a variable blade twist concept can be assumed for the compound, the two aircraft would be equivalent in DOC.

4. The prescribed design noise goal is equivalent to a mean of the community acceptance criteria for selected heliport locations, based on an A-weighted sound measurement corrected for event duration and frequency.

5. The requirement for a speech interference level within the cabin no greater than 70 dB PSIL, equivalent to current fixed wing shorthaul practice, has a significant effect on aircraft design, necessitating transmission acoustic isolation, cabin wall soundproofing, and in the case of the compound, careful selection of auxiliary propulsion.

6. The predicted gross weights for the helicopter and compound 100-passenger designs do not represent unacceptable technical risk associated with size. Experience suggests that growth in weight by more than about 2.5 times the gross weight of the previous largest aircraft of that configuration does engender such risk, unless sufficient weight contingency is included to accommodate it.

7. The rotary wing VTOL offers excellent low speed handling qualities in that 100% control power can be exercised about any axis with little or no reduction in available control power about any orthogonal axis.

8. The requirement for hover out of ground effect with one engine inoperative provides a power margin for safe recovery at any point during a typical take-off procedure following a single engine malfunction.

9. Helicopter external noise can be reduced by 5 dB through moderate reduction in rotor tipspeeds and through adoption of twin low-disc-loading tail rotors, for 4% increase in DOC.

10. Compound external noise can be reduced by 5 dB through moderate reduction in main rotor tipspeed and through adoption of a fan-in-fin anti-torque device, for 6% increase in DOC.

11. For constant take-off gross weight, the baseline helicopter DOC increases by 9% when stagelength is trended out to 740 kilometers (400 nautical miles); the compound by 18%. At short ranges, the helicopter is significantly more economical to operate.

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12. For long-bodied configurations enforced by large main rotor diameters, 6-abreast single-aisle seating is preferred over 7- or 8-abreast dual-aisle arrangements for the same number of passengers.

13. The helicopter satisfies the requirements for gust insensitivity at altitudes up to 3050 meters (10,000 feet). The compound is marginal ir this respect and may require some form of automatic collective aileron control in response to measured normal accelerations.

14. If fuel costs continue to increase, the DOC trend would indicate an advantage in reducing compound cruise speed from 129 m/sec (250 knots) to 118 m/sec (230 knots). The helicopter should maintain its design cruise speed of 89 m/sec (173 knots) even if fuel cost should increase by 200% over the assumed value of 13 cents per gallon.

8.0 RECOMMENDATIONS

1. General comparison of competitive VTOL configurations can most realistically be achieved if timeframe is included as one of the study variables. It is recommended that the results of this and parallel studies be expanded through additional work to include timeframe variation from 1975 to 1990.

2. As indicated in the earlier sections of this report, compound rotor design for low noise and good performance in hover is not compatible with the design requirements for high speed flight. An extension to this study is recommended to relax the groundrule of constant blade geometry to include variable twist and variable compound rotor diameter concepts. The Telescoping Rotor Aircraft (TRAC) rotor system is currently under development at Sikorsky, under contract from the U. S. Army.

3. Because of the anticipated emphasis on fuel economy during the years ahead, it is recommended that the influence on design and operating techniques be assessed as a function of fuel cost and availability.

4. It is recommended that the Advancing Blade Concept (ABC) rotor, currently under development at Sikorsky under Army contract, should be included as a candidate commercial VTOL lift system in a general configuration comparison study including timeframe as a variable.

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